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MX-774
GROUND TO GROUND MISSILE

ACTIVITY

CVAC REPORT 1496-14
26 DECEMBER 1947

CONSOLIDATED VULTEE
SAN DIEGO DIVISION, SAN DIEGO, CALIFORNIA

SECRET

Wood

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MX-774

FOREWORD

PROJECT MX-774, ORIGINALLY A STUDY AND RESEARCH PROGRAM LEADING TO THE PRACTICAL DESIGN OF A SUPERSONIC-SPEED GUIDED MISSILE OF 1500 TO 5000 MILES RANGE, HAS NOW BEEN LIMITED TO THE DESIGN, FABRICATION AND FLIGHT TESTING OF SUPERSONIC TEST VEHICLES. A LIMITED AMOUNT OF WORK IS CONTINUING ON STUDY AND DEVELOPMENT OF LONG-RANGE GUIDANCE SYSTEMS AND STUDIES OF HIGH VELOCITY WARHEAD TEMPERATURES.

THIS REPORT, THE FOURTEENTH IN THE SERIES, SUMMARIZES PROGRESS MADE DURING SEPTEMBER, OCTOBER, AND NOVEMBER 1947 IN MISSILE DESIGN, IN STABILIZATION AND GUIDANCE DEVELOPMENTS, AND IN PREPARATIONS FOR TESTING THE MX-774 VEHICLES BEING FABRICATED.

REPORT CONTENTS

ILLUSTRATIONS

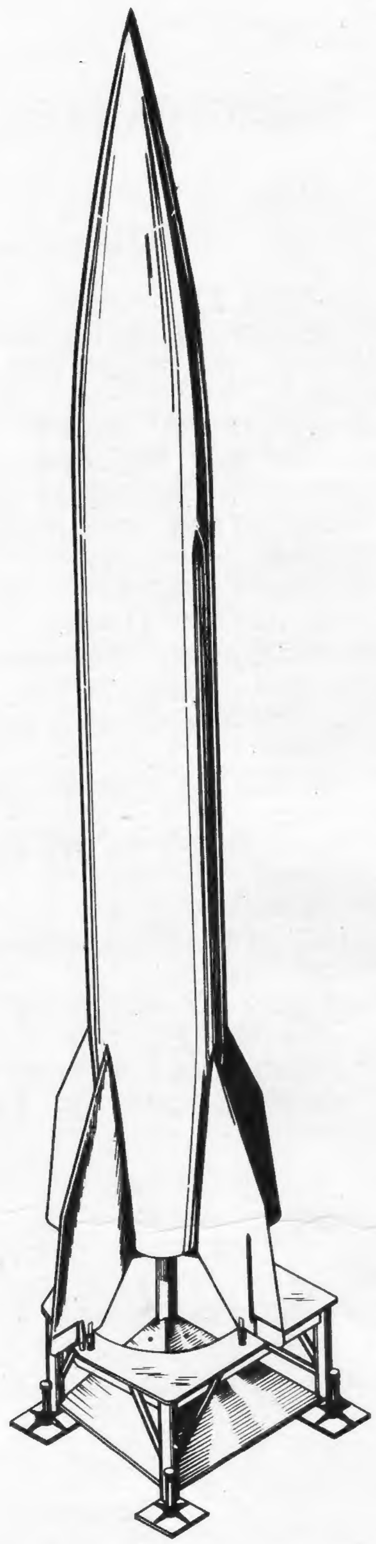


FIG I -- MX-774B FLIGHT TEST MISSILE ON LAUNCHING PAD

SECTION I
SUMMARY

A. PROBLEMS INITIATED - WORK CONDUCTED

A GENERAL DESCRIPTION OF THE MX-774B FLIGHT TEST VEHICLE AND AN OUTLINE OF STABILIZATION AND GUIDANCE PROVISIONS FOR THE MX-774 MISSILE ARE CARRIED IN SECTION II TO SUPPLY A FRAME FOR THE PICTURE OF RECENT DEVELOPMENT WORK ON THE PROJECT.

MINOR CHANGES IN ACTUAL WEIGHTS HAS MADE DESIRABLE THE RE-PRESENTATION IN SECTION III OF CG TRAVEL WITH FUEL CONSUMPTION AND OF WEIGHT FABRICATED. ASSEMBLY OF THE FLIGHT TEST VEHICLE IS PROGRESSING RAPIDLY. STRESS ANALYSIS ON THIS VEHICLE IS SUMMARIZED.

ANALYTICAL INVESTIGATIONS, SECTION IV, ROUND OUT INFORMATION ON (1) ADAPTATION OF THE MX-774 TO ATTAIN MAXIMUM ALTITUDES OF 500,000 TO 1,000,000 FEET, (2) FLIGHT PATTERN OF THE FLIGHT TEST VEHICLE, (3) OPTIMUM SERVO CONSTANTS FOR MISSILE STABILIZATION, (4) DISSOCIATION REACTIONS OF ATMOSPHERIC GASES AS INFLUENCED BY BOUNDARY LAYER PRESSURES, AND (5) INCREASE OF ROCKET THRUST FROM USE OF EXPENDABLE NOZZLE AREA REDUCERS.

DAMAGES TO SOME OF THE ENGINE SECTION COMPONENTS, SUSTAINED WHEN TWO ROCKET CYLINDERS BLEW UP DURING AN ATTEMPTED FIRING RUN, ARE PICTURED IN SECTION V.

STABILIZATION SYSTEM EQUIPMENT WAS INSTALLED IN THE STATIC TEST MISSILE PRIOR TO THE STATIC FIRING TEST. THE NO. 2 STABILIZATION PANEL ASSEMBLY IS BEING COMPLETED FOR THE FLIGHT TEST VEHICLE. THE COMMAND CONTROL DRIVE UNIT FOR THIS SAME VEHICLE IS COMPLETED AND UNDERGOING LABORATORY TESTS. TEST EQUIPMENT IS ITEMIZED IN SECTION VI.

THE POSITION TRACKER GROUND STATION EQUIPMENT HAS BEEN RACKED READY FOR USE. A FIELD TEST OF THE PHASE COMPARISON TRACKING METHOD IS DESCRIBED IN SECTION VII. A QUARTER INCH MOVEMENT AT 240 FEET WAS EASILY DETECTABLE; MORE REFINED TESTS ARE PLANNED.

Truck installation of telemetering was accomplished and airborne telemetering components placed in the Static Test Missile before the firing run. The explosion interrupted successful operation of all equipment.

An extensive report of the static firing test is given in Section IX. Satisfactory "cold" checks are detailed and the mishap during the "hot" run is discussed.

A list of internal reports released under project MX-774 may be found in the Reference Section.

B. PROJECT CONTACTS

Visit from Watson Laboratories personnel. A discussion was held concerning technical details of the complete guidance program. An inspection of missile equipment and laboratory facilities was made to enable a detailed report to Wright Field.

Visit from Watson Laboratories personnel. Vacuum tube developments were discussed. Technical information was exchanged in this field.

Visit from Hq. S.A.C. personnel. An inspection of guidance and control equipment for the missile was conducted.

Visit from A.M.C. personnel. An inspection was made of equipment to be used in MX-774 Test Vehicle and a discussion was held concerning the future program.

A demonstration of Azimuth tracking phase comparison equipment performance was made in the presence of Wright Field representatives. Representatives were furnished with general information concerning various developments for guidance and control of the missile.

A demonstration of position tracker system performance was made 25 November with Dr. Marcus O'Day of Cambridge Field Station present. Information requested by Dr. O'Day was forwarded to him.

SECTION II
DESCRIPTION OF MISSILE

A. GENERAL

THE MX-774 TEST VEHICLE IS A SELF-LAUNCHED, HIGH-ALTITUDE SUPERSONIC ROCKET PROJECTILE PROPELLED BY AN ALCOHOL-OXYGEN MOTOR OF 8000-POUNDS THRUST. IT IS BEING DEVELOPED TO STUDY POWER PLANT, CONTROL, GUIDANCE, AERODYNAMIC, AND TRAJECTORY PROBLEMS. THE MX-774B INBOARD PROFILE IS SHOWN IN FIGURE 2.

B. PRINCIPAL DIMENSIONS AND PARTICULARS

GENERAL		
SPAN	82.24	INCHES
LENGTH (OVERALL)	379	INCHES
HEIGHT AND WIDTH (MAXIMUM)	82.24	INCHES
GROSS WEIGHT (INITIAL)	4,216	POUNDS
GROSS WEIGHT (END OF BURNING)	1,301	POUNDS
ROCKET THRUST	8,000	POUNDS
FINS		
AIRFOIL SECTION	MODIFIED DOUBLE WEDGE	
ROOT CHORD	69	INCHES
OVERALL CHORD	85	INCHES
INCIDENCE	0	DEGREES
DIHEDRAL	0	DEGREES
SWEEPBACK	60	DEGREES
AREA (PER FIN)	10.02	SQ FT
TAB AREA (PER TAB)	0.48	SQ FT
FUSELAGE		
DIAMETER	30	INCHES
LENGTH	362	INCHES
MAXIMUM SECTIONAL AREA	4.91	SQ FT
ROCKET CYLINDER RANGE OF MOVEMENT		
PITCH CONTROL	5	DEGREES
ROLL CONTROL	5	DEGREES
YAW CONTROL	5	DEGREES
TANK CAPACITIES		
FUEL - ALCOHOL	196	GALLONS
OXIDIZER - LIQUID OXYGEN	167	GALLONS
NITROGEN - PROPELLANT TANK	4,140	CU IN.
PRESSURIZING	1,090	PSI
NITROGEN - H_2O_2 TANK	794	CU IN.
PRESSURIZING	2,000	PSI
H_2O_2 TANK	5.3	GALLONS

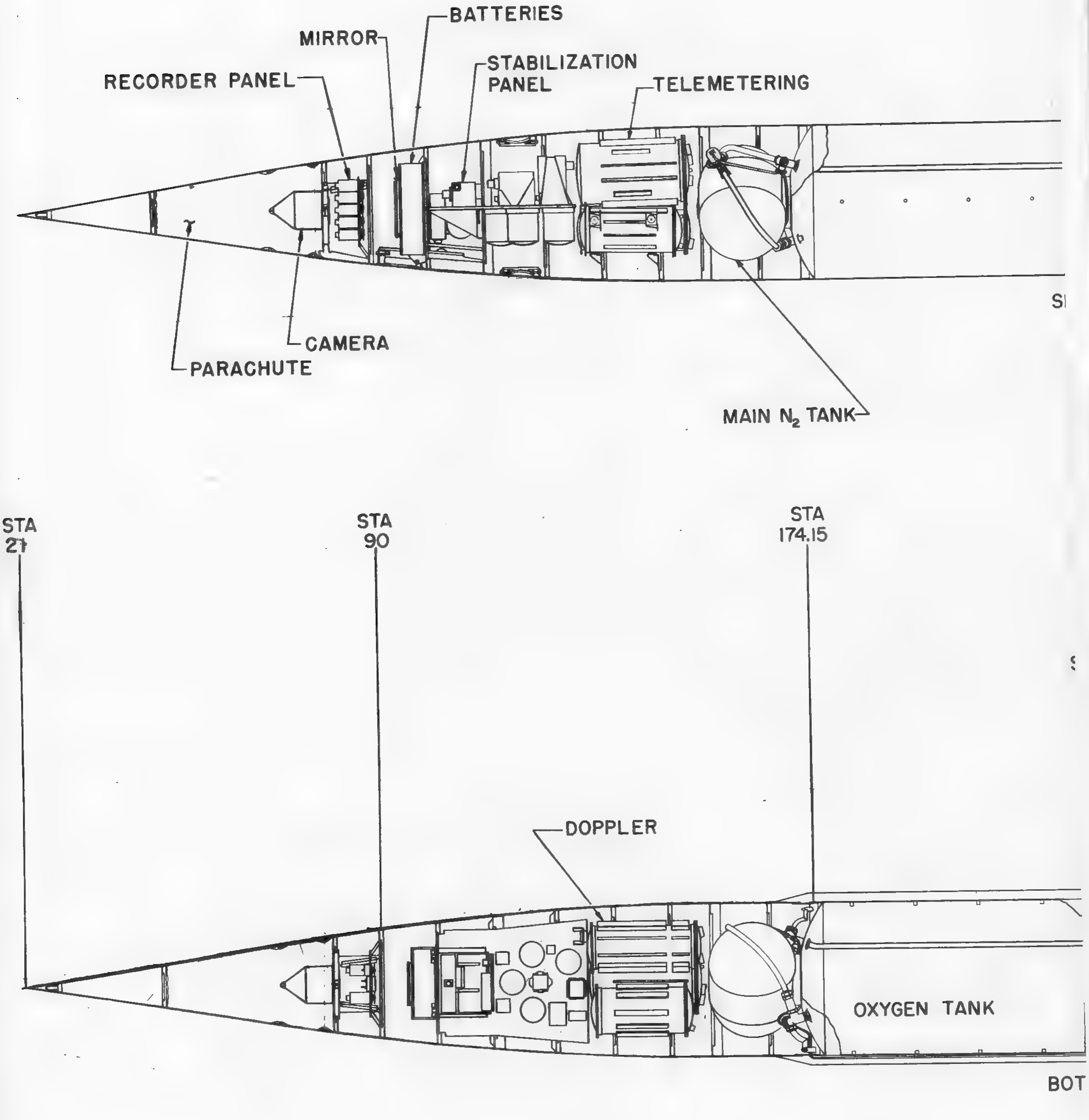
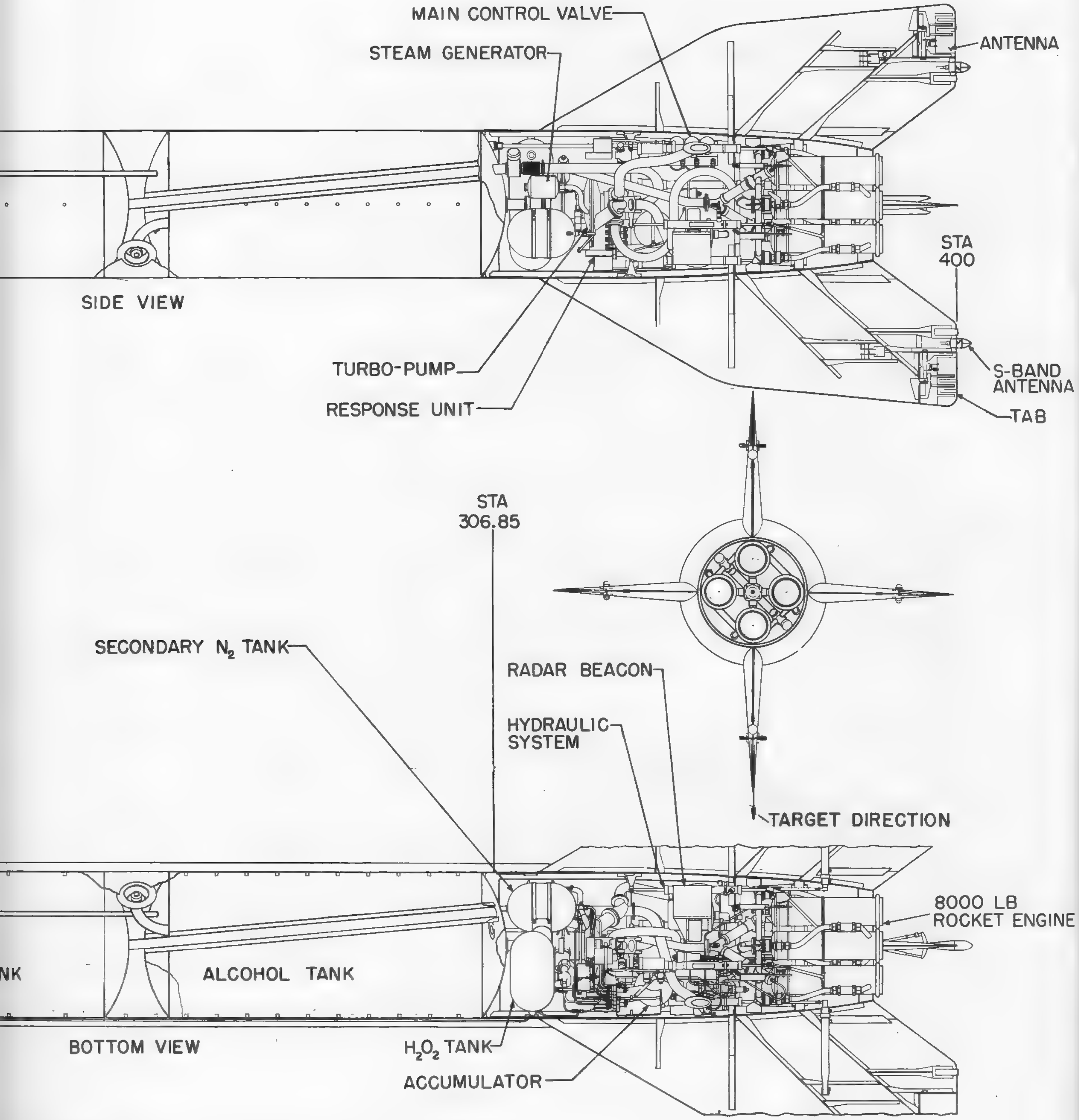


FIG 2 -- INBOARD PROFI



ARD PROFILE - MX-774B FLIGHT TEST VEHICLE

6002-000003

C. AIRFRAME

1. FUSELAGE

THE FUSELAGE IS COMPRISED OF A REMOVABLE TAIL CONE, A TAIL ASSEMBLY INCLUDING THE FINS, A CENTER SECTION COMPOSED OF INTEGRAL PROPELLANT TANKS, A NOSE ASSEMBLY, AND A NOSE CONE.

THE ALCOHOL AND LIQUID-OXYGEN TANKS FORMING THE CENTER SECTION ARE OF WELDED ALUMINUM ALLOY CONSTRUCTION. TWO FAIRINGS WHICH HOUSE WIRES AND PRESSURE LINES ARE ATTACHED TO THE TANKS AND MAY BE REMOVED. ACCESS TO THE COMPARTMENT BETWEEN THE TANKS IS BY MEANS OF A REMOVABLE PANEL. THE ALCOHOL TANK IS FILLED THROUGH A FILLER NECK LOCATED BETWEEN THE TWO TANKS. THE LIQUID-OXYGEN TANK IS FILLED THROUGH A FILLER VALVE LOCATED AT THE BOTTOM OF THE CENTER SECTION.

2. TAIL AND EMPENNAGE

THE TAIL GROUP CONSISTS OF FOUR FULL-CANTILEVER SURFACES, EACH CONTAINING A CONTROLLED TAB. BASIC STRUCTURE CONSISTS OF SPANWISE FABRICATED SHEET ALUMINUM ALLOY RIBS AND SPARS COVERED WITH ALUMINUM ALLOY SKIN.

ONE PAIR OF TABS PERFORMS A SLOW TRIM ROLL CORRECTION AND ARE ACTUATED THROUGH FLEXIBLE SHAFTS BY A LEAR ACTUATOR TO ATTAIN FULL 20-DEGREE DEFLECTION IN EITHER DIRECTION IN ELEVEN SECONDS. LIMIT SWITCHES ON THE ROLL MASTER HYDRAULIC CYLINDER ENERGIZE THE SLOW TRIM TAB ACTUATOR WHEN THE SWIVELING ROCKET MOTORS HAVE ATTAINED 1-DEGREE ROLL DEFLECTION.

THE SECOND PAIR OF TABS ARE CONNECTED THROUGH A TORQUE TUBE AND LINKAGE TO A CORRESPONDING PAIR OF ROCKET CYLINDERS AND ARE DEFLECTED THROUGH 20 DEGREES IN EITHER DIRECTION BY 10 DEGREES DEFLECTION OF THE ROCKET CYLINDER.

THE TAIL CONE IS AN ALUMINUM ALLOY STRUCTURE WHICH MAY BE REMOVED WITHOUT ANY DISTURBANCE OF MECHANICAL PARTS TO PROVIDE FOR ENGINE ACCESS AND REMOVAL. THE PRINCIPAL FUNCTION OF THE TAIL CONE IS TO ACT AS A FAIRING.

THE EMPENNAGE ASSEMBLY INCLUDES THE TAIL ASSEMBLY, WHICH IS AN ALUMINUM ALLOY SEMI-MONOCOQUE STRUCTURE ACTING AS THE HOUSING AND SUPPORT FOR THE POWER PLANT, AND THE FINS. ACCESS DOORS, CONSTITUTING ONE-QUARTER OF THE TAIL ASSEMBLY SKIN, ARE PROVIDED. THE EMPENNAGE ASSEMBLY, FIGURE 3, INCLUDING THE POWER PLANT COMPONENTS, CAN BE REMOVED FROM THE MAIN

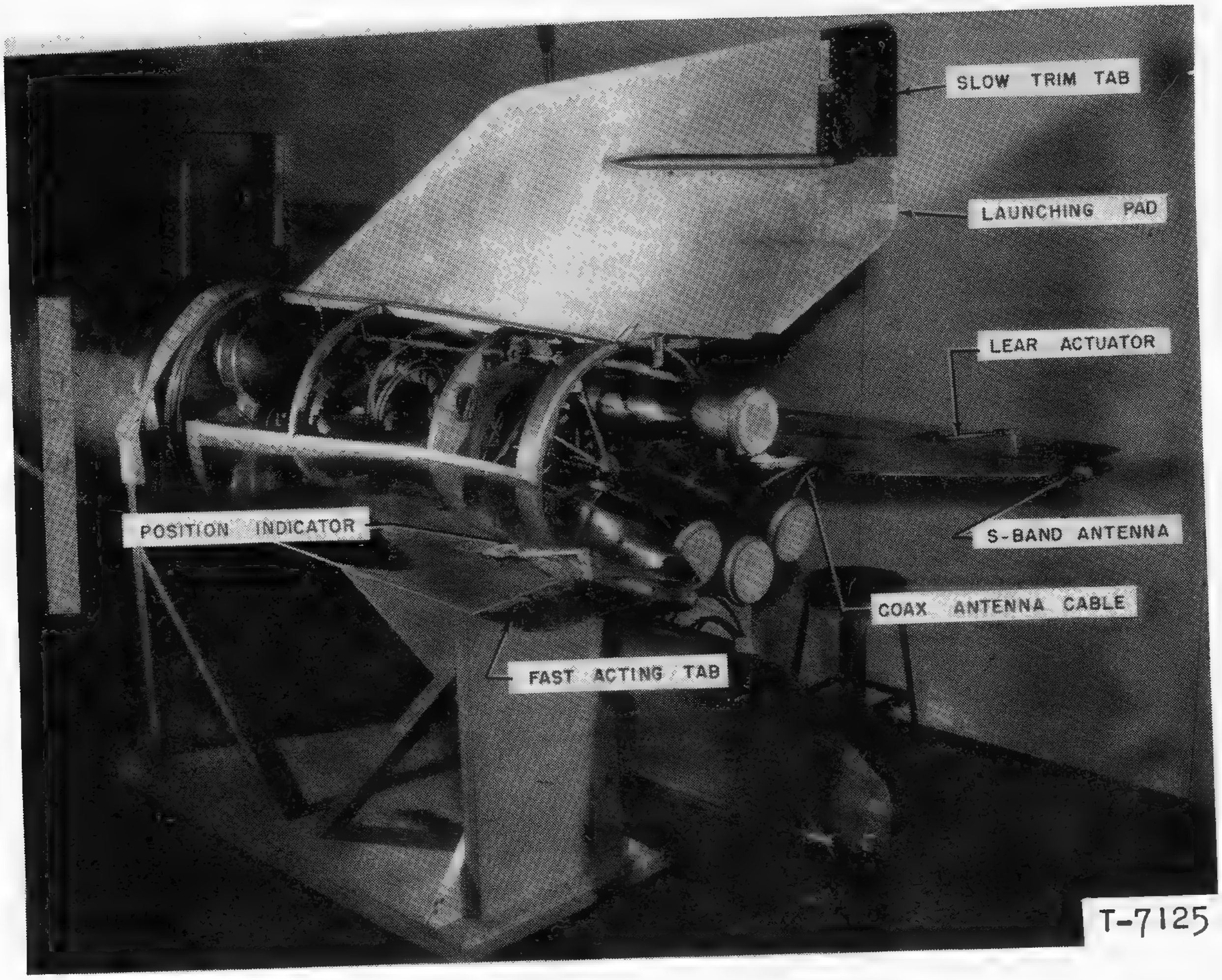


FIG 3 -- MX-774B EMPENNAGE ASSEMBLY - TAIL CONE REMOVED

FUSELAGE ASSEMBLY AFTER REMOVAL OF THE FAIRINGS AND DISCONNECTION AT THE ALCOHOL LINE FLANGE, THE OXYGEN LINE FLANGE, THE CONTROL CYLINDER ALCOHOL RETURN LINE FITTING, THE NITROGEN PRESSURE LINE FITTING, AND THE ELECTRICAL DISCONNECTS IN THE FAIRING TUNNELS.

3. Nose

THE NOSE CONE IS AN ALUMINUM ALLOY SEMI-MONOCOQUE STRUCTURE WHICH HOUSES A 40-FOOT RIBBON TYPE PARACHUTE AND IS SEVERED FROM THE MISSILE DURING FLIGHT BY MEANS OF AN ELECTRICALLY DETONATED "PRIMA-CHORD".

THE NOSE ASSEMBLY IS OF ALUMINUM ALLOY SEMI-MONOCOQUE CONSTRUCTION AND HOUSES THE INTELLIGENCE AND GUIDANCE EQUIPMENT AND PAY LOAD OF THE MISSILE. IT MAY BE REMOVED FROM THE TANK SECTION AFTER REMOVAL OF THE FAIRING.

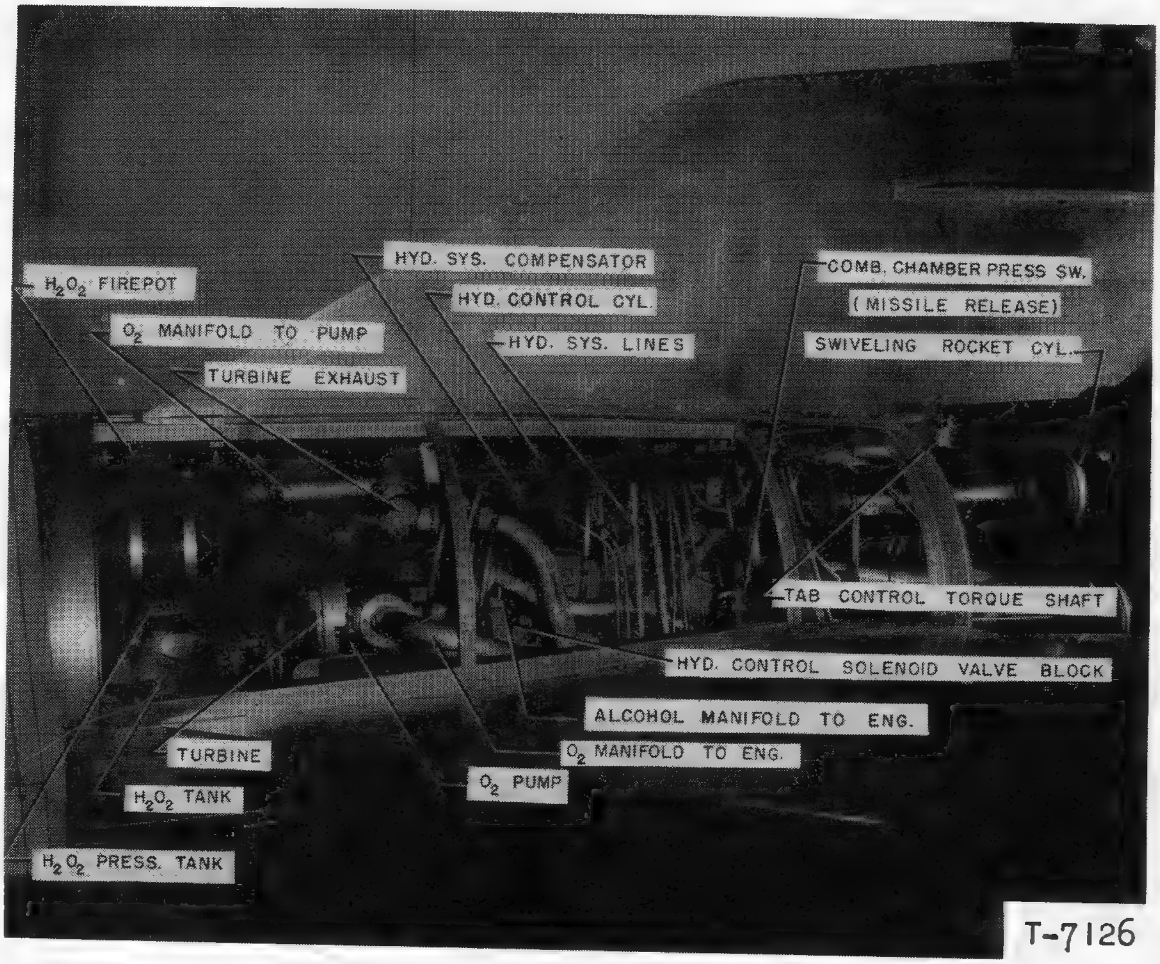


FIG 4 -- MX-774B POWER PLANT INSTALLATION

D. PROPULSION SYSTEM

THE POWER PLANT INSTALLATION, FIGURE 4, COMPRISES A REACTION MOTORS INCORPORATED ALCOHOL-OXYGEN BURNING ROCKET ENGINE MOUNTED IN A TUBULAR STEEL MOUNT AND A FUEL AND OXIDIZER SYSTEM THROUGH WHICH ALCOHOL AND OXYGEN ARE SUPPLIED TO THE ROCKET CYLINDERS AT 450 PSI PRESSURE BY A TURBO-PUMP DRIVEN BY THE PRODUCTS OF DISSOCIATION OF H_2O_2 WHEN ACTED UPON BY A CATYLIST. PROPELLANT TANKS ARE MAINTAINED UNDER A CONSTANT PRESSURE BY MEANS OF PRESSURE REGULATED NITROGEN.

THE 8000-C4 ENGINE CONSISTS OF FOUR 2000-POUND THRUST ROCKET CYLINDERS SWIVEL MOUNTED TO A TUBULAR STEEL MOUNT WHICH ALSO SERVES AS THE ALCOHOL MANIFOLD. MOUNTED IN THE MANIFOLD

ARE TWO RMI PROPELLANT VALVES ADMITTING THE FLOW FROM THE ALCOHOL PUMP. OXYGEN IS FED DIRECTLY FROM AN OXYGEN PROPELLANT VALVE THROUGH A SWIVELING JOINT AND CHECK VALVE TO EACH CYLINDER. THE CYLINDERS ARE SWIVELED PLUS OR MINUS 10 DEGREES FROM NEUTRAL POSITION BY MEANS OF HYDRAULIC ACTUATING CYLINDERS.

E. MISSILE STABILIZATION

STABILIZATION INCORPORATES A COMBINATION ELECTRICAL AND HYDRAULIC CONTROL SYSTEM. EFFECTS OF A MISSILE PITCH, ROLL, OR YAW ON THE STABILIZATION GYROS ARE FIRST CONVERTED INTO AN ELECTRICAL ERROR SIGNAL WHICH IS PROPORTIONAL IN SIGN AND MAGNITUDE TO THE MISSILE DISPLACEMENT AND RATE OF DISPLACEMENT. THE STABILIZATION SYSTEM ALSO INCLUDES INTEGRAL CONTROL UNITS FOR PITCH, ROLL AND YAW TO CORRECT FOR MANUFACTURING MISALIGNMENTS OR UNBALANCED CONDITIONS WHICH MAY OCCUR. THE ELECTRICAL ERROR SIGNALS FROM THESE COMPONENTS AND FROM ROCKET POSITION FEED-BACKS ARE MIXED, AMPLIFIED, AND CONVERTED INTO PROPORTIONAL HYDRAULIC VALVE DISPLACEMENTS. VALVE ACTIONS ARE FOLLOWED BY INDIVIDUAL POWER CYLINDER MOVEMENTS WHICH CHANGE THE ANGULAR POSITIONS OF THE FOUR ROCKET CYLINDERS TO CORRECT THE MISSILE'S FLIGHT PATH.

POWER FOR OPERATING THE HYDRAULIC CYLINDERS IS SUPPLIED BY A POWER CYLINDER FOR EACH OF THE ROLL, YAW, AND PITCH SYSTEMS. THE SYSTEM OPERATES ON ALCOHOL PRESSURE FROM THE FUEL PUMP DISCHARGE. THE ALCOHOL FLOWS FROM THE DISCHARGE LINE THROUGH AN ACCUMULATOR AND FILTER TO A SOLENOID OPERATED CONTROL VALVE WHICH, ON RECEIVING A SIGNAL FROM THE ELECTRONIC GUIDANCE SYSTEM, REGULATES THE FLOW TO THE ROLL, YAW, OR PITCH POWER CYLINDER. THE POWER CYLINDER IMPARTS MOTION TO THE ROCKET CYLINDER TO WHICH IT IS CONNECTED AND, THROUGH THE HYDRAULIC SLAVE CYLINDERS, TO THE OTHER ROCKET CYLINDERS TO BE SWIVELED BY THE SIGNAL RECEIVED. A COMPENSATOR IS INSTALLED IN EACH OF THE ROLL, YAW AND PITCH SLAVE SYSTEMS TO COMPENSATE FOR TEMPERATURE CHANGES AND LEAKAGE LOSSES.

FLIGHT CONTROL OF RANGE IS ACCOMPLISHED BY EITHER A MISSILE BORNE PROGRAM UNIT OR A GROUND BASED COMMAND THAT SHIFTS THE REFERENCE COORDINATES OF THE STABILIZATION SYSTEM AND SHUTS OFF THE POWER WHEN THE DESIRED VELOCITY IS ATTAINED.

F. ELECTRONIC GUIDANCE

THE GUIDANCE SYSTEM, FIGURE 5, FOR A LONG RANGE MX-774 UTILIZES THREE SYSTEMS FOR DETERMINATION OF POSITION, SPEED, AND RANGE OF THE MISSILE.

(1) A VHF POSITION TRACKER SYSTEM SERVES AS A PRECISION DIRECTION FINDER IN BOTH AZIMUTH AND ELEVATION BY UTILIZING PHASE COMPARISONS OF WAVE-FRONT ARRIVALS. THE REQUISITE SIGNALS ARE TRANSMITTED FROM A BEACON IN THE MISSILE.

(2) A DOPPLER SPEEDOMETER SYSTEM MEASURES VELOCITY BY FREQUENCY COMPARISON OF RADIO SIGNALS SENT TO THE MISSILE WITH SIGNALS RETURNED FROM A FREQUENCY DOUBLING REPEATER STATION LOCATED IN THE MISSILE.

(3) A RANGING SYSTEM MEASURES RANGE BY PHASE COMPARISON OF SIGNALS RETURNED FROM THE SAME REPEATER STATION.

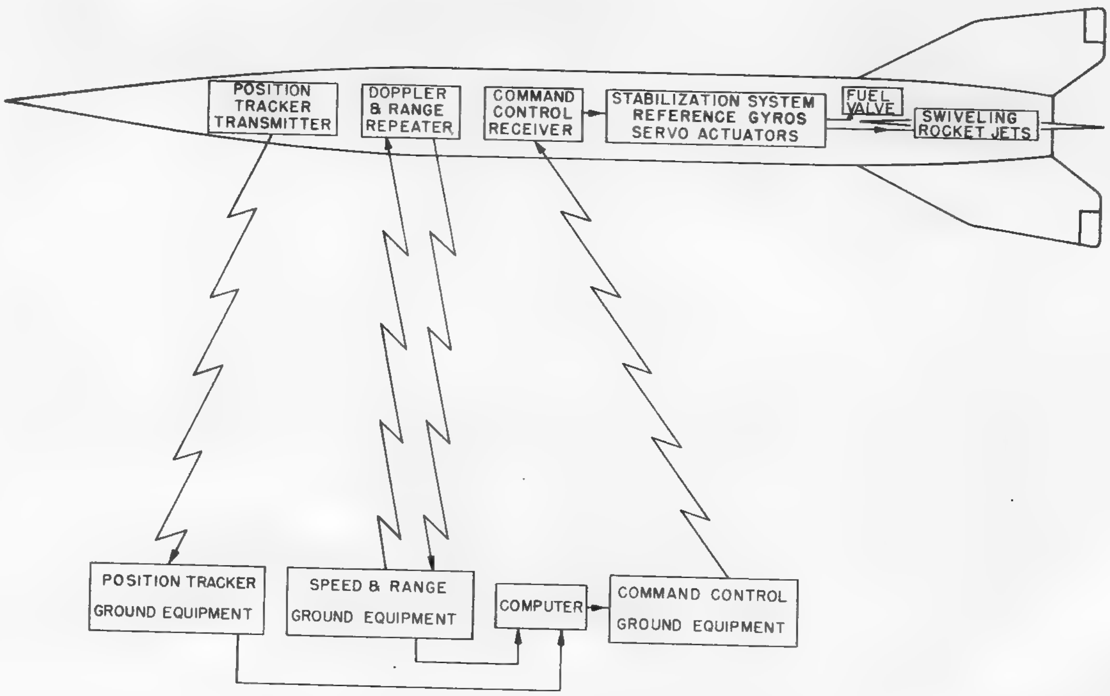


FIG 5 -- SIMPLIFIED BLOCK DIAGRAM OF MX-774 GUIDANCE CONTROL AND STABILIZATION

SPEED, DIRECTION, AND DISTANCE INTELLIGENCE FROM THE ABOVE UNITS IS FORWARDED TO A COMPUTER WHICH, IN A HIGH-SPEED PRECISION OPERATION, COMPARES THE INTEGRATED DATA WITH A PRE-COMPUTED FLIGHT PATH. CORRECTIONS ARE THEN MADE BY THE COMPUTER WHICH, WHEN APPLIED THROUGH A COMMAND CONTROL RADIO LINK, RETURNS THE MISSILE TO A PATH CONVERGING WITH THE PRE-COMPUTED PATH.

THE COMMAND CONTROL SYSTEM DIRECTS THE MISSILE IN AZIMUTH AND ELEVATION AND PROVIDES FUEL SHUT-OFF. THE GROUND TRANSMITTER RADIATES INTELLIGENCE SIGNALS CONSISTING OF A PHASE DIFFERENCE BETWEEN TWO C-W CARRIERS. THE PHASE SHIFT IS PROPORTIONAL TO THE MAGNITUDE OF THE INTELLIGENCE TO BE TRANSMITTED. THE AIRBORNE RECEIVER AND ITS ASSOCIATED EQUIPMENT RECEIVES AND INTERPRETS THE INTELLIGENCE SIGNALS AND DELIVERS THE CONTROL SIGNAL TO THE PROPER FUNCTION BY MEANS OF THE MISSILE SERVO SYSTEM.

SECTION III
DESIGN AND FABRICATION

A. GENERAL

CG TRAVEL IN THE CURRENT DESIGN FOR THE MX-774 FLIGHT TEST ARTICLE IS SHOWN IN FIGURE 6. THE WEIGHT STATUS IS REPORTED IN FIGURE 7. CONTRACTOR CONTROLLED WEIGHT IS 622 POUNDS AS COMPARED TO 613 POUNDS IN THE PREVIOUS REPORT (REFERENCE 2). THIS DIFFERENCE IS ACCOUNTED FOR BY MISCELLANEOUS WEIGHT CHANGES AND BY AN 8-POUND TRANSFER FROM CONTRACTOR PURCHASED EQUIPMENT FOR THE REVISED BATTERY SETUP.

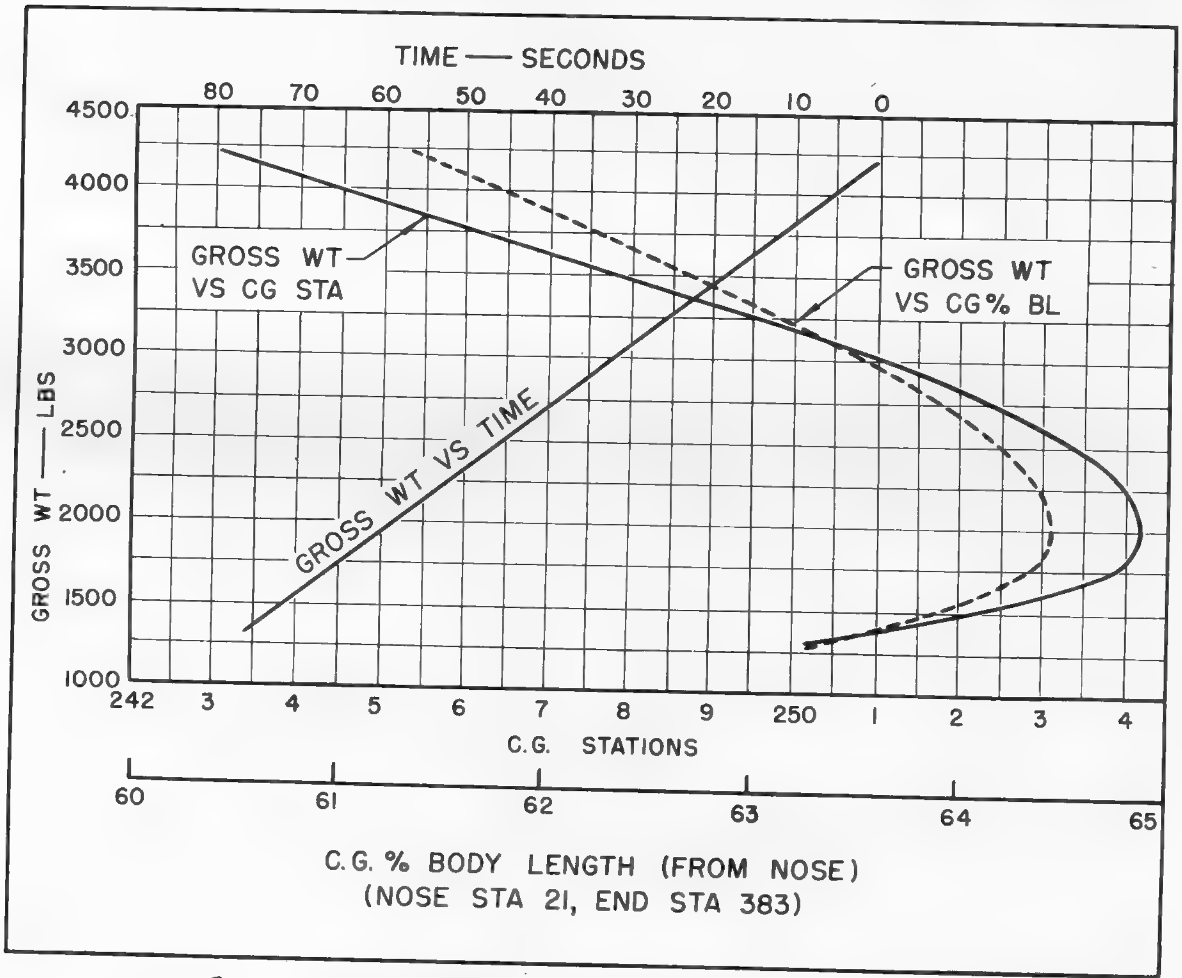


FIG 6 -- GROSS WEIGHT VERSUS CG AND TIME FOR MX-774B

ITEM	PREVIOUS REPORT		PRESENT REPORT		CHANGES
WEIGHT EMPTY	(1296.0)		(1301.0)		(+5.0)
BODY GROUP	238.0		239.0		1.0
FIN GROUP	104.0		104.0		
POWER PLANT GROUP	426.0		426.0		
FIXED EQUIP. GROUP	302.0		304.0		+2.0
FLIGHT TEST EQUIP.	226.0		228.0		+2.0
USEFUL LOAD	(2915.0)		(2915.0)		(0)
ALCOHOL	1350.0		1350.0		
OXYGEN	1500.0		1500.0		
HYDROGEN PEROXIDE	65.0		65.0		
GROSS WEIGHT	4211.0		4216.0		+5.0
CG IN % B.L., FROM NOSE					
WEIGHT EMPTY	62.9		63.3		+0.4
GROSS WEIGHT	61.2		61.3		+0.1
STATUS OF WEIGHT DATA	LB	%	LB	%	
WEIGHT EMPTY	(1296)		(1301)		
ESTIMATED	204	16	84	6	
CALCULATED	326	25	245	19	
ACTUAL	766	59	972	75	
CONTRACTOR FABRIC. WGT.	(613)		(622)		
ESTIMATED	51	8	11	2	
CALCULATED	167	27	155	25	
ACTUAL	395	65	456	73	

EXPLANATION OF CHANGES	CHANGES POUNDS
WEIGHT EMPTY	
BODY GROUP - ELECTRONIC EQUIPMENT ACCESS DOOR ADDED	+ 1.0
FIXED EQUIPMENT GROUP	
GUIDANCE INSTAL. - REVISED	+ 0.8
ANTENNA - S BAND. - REVISED	+ 0.2
DESTRUCTION CHARGE ADDED	+ 1.3
ENGINE SWIVELING CONTROLS - ACTUALS	− 0.3
FLIGHT TEST EQUIPMENT	
DUAL TEMPERATURE INDICATOR ADDED	+ 2.0

FIG 7 -- WEIGHT STATUS TABLE

B. FLIGHT TEST VEHICLE STATUS

THE LAST OF THE ENGINEERING DRAWINGS FOR THE FLIGHT TEST VEHICLE WERE RELEASED IN SEPTEMBER. THESE DRAWINGS DETAILED THE CAMERA CONTAINER, NEW BATTERY PROVISIONS, RADAR BEACON MOUNTING, REVISED NOSE STRUCTURE, AND ELECTRONIC COMPONENT CONTAINERS AND SUPPORTS.

FORWARD NOSE, AFT NOSE, EMPENNAGE-FIN-TAB, AND TAIL CONE SECTIONS HAVE BEEN COMPLETED. TANK FAIRINGS ARE IN WORK. INSTALLATION OF THE TAB CONTROL ACTUATORS AND LINKAGES HAS BEEN STARTED. OXYGEN, ALCOHOL, AND SECONDARY NITROGEN TANKS ARE FINISHED. HYDROGEN PEROXIDE, AND MAIN NITROGEN TANK DESIGNS HAVE BEEN MODIFIED AND ARE BEING FABRICATED.

A PROPOSAL HAS BEEN FORMULATED DESCRIBING THE PARACHUTE TO BE INSTALLED IN THE FORWARD NOSE SECTION. PARACHUTE VENDORS ARE BEING ASKED TO PRESENT BIDS.

PHOTO-RECORDER PANEL, STABILIZATION PANEL, TELEMETERING ASSEMBLY, AND DOPPLER VELOCITY AND RANGE EQUIPMENT ARE BEING READIED FOR INSTALLATION. THE RADAR BEACON IS NEARING COMPLETION. ANTENNA INSTALLATION IS PARTIALLY FINISHED.

ELECTRICAL WIRING AND HYDRAULIC PLUMBING INSTALLATIONS ARE PROGRESSING.

THE ROCKET ENGINE IS DUE IN FROM REACTION MOTORS INCORPORATED BY FEBRUARY 1948. PROPULSION SYSTEM PLUMBING IS READY FOR INSTALLATION EXCEPT FOR CERTAIN RMI SUPPLIED PARTS.

C. STATIC FIRING TEST MISSILE STATUS

THE STATIC FIRING TEST MISSILE HAS BEEN RETURNED TO THE SHOPS TO BE REWORKED AND REWIRED. NON-ESSENTIAL PARTS MAY BE LEFT OFF THE MISSILE FOR THE NEXT TEST RUNS. MUCH OF THE SMALL INSTRUMENTATION WILL BE REPLACED.

D. STRESS ANALYSIS OF THE FLIGHT TEST MISSILE

THE REPORT ON STRESS ANALYSIS OF THE MX-774B (REFERENCE 1) HAS BEEN COMPLETED. PART I IS A COMPUTATION OF BODY LOADS, SHEARS, AND BENDING MOMENTS. PART II IS A DETAILED STRESS ANALYSIS OF THE VARIOUS COMPONENTS OF THE MISSILE.

THE INVESTIGATION OF LOADS TO BE USED FOR STRUCTURAL DESIGN OF THE MX-774B INCLUDED CONSIDERATION OF THOSE LOADS PRODUCED IN THE MANEUVERING OF THE MISSILE AS WELL AS THOSE RESULTING FROM THE MISSILE ENCOUNTERING A GUST.

IT WAS ASSUMED THAT THE MOST SEVERE MANEUVERING LOADS WOULD OCCUR WITH INSTANTANEOUS FULL DEFLECTION OF BOTH FIN TABS AND DRIVING ROCKETS AND THAT THIS CONDITION COULD OBTAIN AT ANY POINT ON THE FLIGHT PATH UP TO THE END OF BURNING OF THE ROCKETS. IT WAS FOUND THAT DEFLECTION OF THE TABS AND ROCKETS PRODUCED AN OSCILLATORY MOTION ABOUT THE CENTER OF GRAVITY OF THE MISSILE WHICH AT MAXIMUM AMPLITUDE CAUSED A RELATIVELY LARGE ANGLE OF ATTACK.

AERODYNAMIC FORCES ON THE MISSILE RESULTING FROM ENCOUNTERING A GUST HAVING A VELOCITY OF 75 FEET PER SECOND OR $50/\sqrt{\sigma}$ (WHICHEVER IS HIGHER) PRODUCED LOADS WHICH WERE LESS SEVERE THAN THOSE OBTAINED IN THE MANEUVER CONDITION; HENCE, COMPUTATIONS FOR THE GUST CONDITION WERE NOT CARRIED TO COMPLETION.

CRITICAL MANEUVERING LOADS WERE FOUND TO BE OBTAINED ON THE VERTICAL FLIGHT PATH AT AN ALTITUDE OF 16,900 FEET, AT WHICH POINT THE MISSILE IS TRAVELING AT A MACH NUMBER OF 1.07.

IN ANALYSING, THE AERODYNAMIC LOADS ON BODY AND FINS WERE DISTRIBUTED TO STATIONS AT TWENTY INCH INTERVALS ALONG THE LENGTH OF THE BODY, USING A THEORETICAL DISTRIBUTION FOR THE BODY LOAD AND A DISTRIBUTION BASED ON GERMAN V-2 WIND TUNNEL TESTS FOR THE FIN LOAD. AIR LOAD SHEARS AND BENDING MOMENTS WERE COMPUTED AT EACH SUCH STATION ALONG THE BODY.

A DETAILED WEIGHT DISTRIBUTION WAS MADE, DISTRIBUTING THE WEIGHT OF THE MISSILE TO STATIONS AT TWENTY-INCH INTERVALS ALONG THE LENGTH OF THE BODY. UNIT SHEARS AND BENDING MOMENTS AT THESE STATIONS DUE TO A ONE "G" ACCELERATION NORMAL TO THE LONGITUDINAL AXIS AND DUE TO A ONE RADIAN PER SECOND SQUARED ANGULAR ACCELERATION WERE COMPUTED.

NET SHEARS AND BENDING MOMENTS IN THE MISSILE BODY WERE CALCULATED BY COMBINING AIR LOAD SHEARS AND BENDING MOMENTS WITH THOSE DUE TO INERTIA AT THE PROPER LOAD FACTORS. NET SHEARS AND BENDING MOMENTS ALONG THE BODY FOR THE MANEUVERING CONDITION IN VERTICAL FLIGHT AT 16,900 FEET ALTITUDE ARE PLOTTED AGAINST BODY STATION IN FIGURE 8.

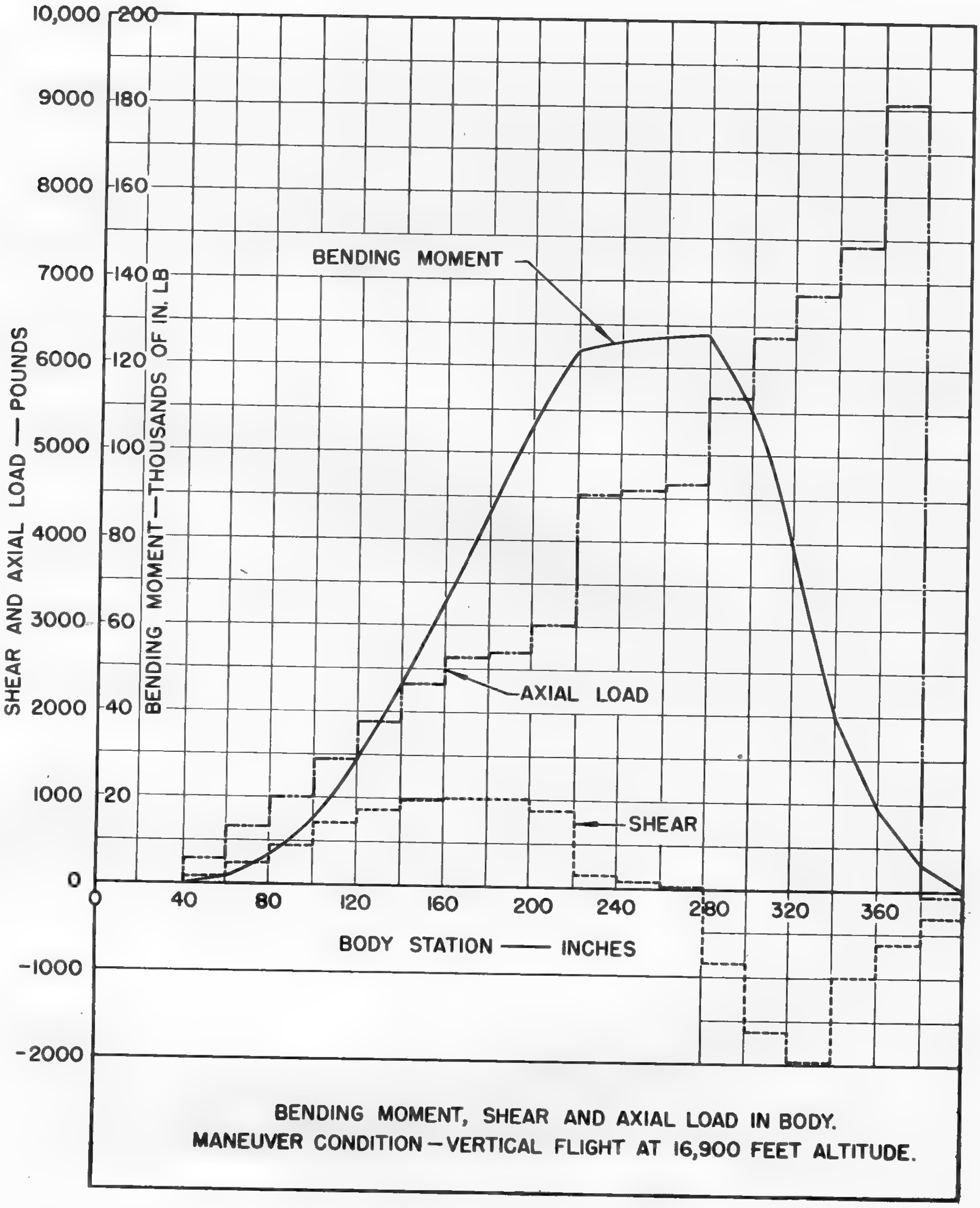


FIG 8 -- NET SHEAR, BENDING MOMENT AND AXIAL LOAD IN MX-774B

IN THE DETERMINATION OF AXIAL LOADS IN THE BODY, AERODYNAMIC DRAG FORCES ON THE BODY AND FINS WERE COMPUTED, AND THE LONGITUDINAL INERTIAL FORCE OBTAINED BY EQUATING THE TOTAL ROCKET THRUST TO THE SUM OF THE DRAG AND INERTIA FORCES. AXIAL LOAD IN THE BODY AT THE VARIOUS STATIONS WAS DETERMINED BY SUMMATION OF THE LONGITUDINAL FORCES ON EITHER SIDE OF EACH STATION. FIGURE 8 SHOWS AXIAL LOAD IN THE BODY PLOTTED AGAINST BODY STATION FOR THE CRITICAL CONDITION.

THE CRITICAL LOAD ON THE TAB WAS FOUND TO BE OBTAINED IN THE GUST CONDITION IN THE MAXIMUM RANGE FLIGHT PATH AND AMOUNTED TO 328 POUNDS.

SECTION IV
DYNAMICS ANALYSES

A. AERODYNAMIC INVESTIGATIONS

1. AS A RESULT OF THE USAF REQUEST FOR INFORMATION ON INCREASED PAYLOAD VOLUMES FOR THE SINGLE-STAGE TEST VEHICLE WITH SPECIAL REFERENCE TO STABILITY LIMITATIONS AND LOSSES IN MAXIMUM ALTITUDE, THE FOLLOWING INVESTIGATIONS WERE MADE.

A. PAYLOAD WEIGHTS NECESSARY TO STABILIZE THE TEST VEHICLE FOR VARIOUS PAYLOAD VOLUMES (FIGURE 9).

B. MAXIMUM ALTITUDE ATTAINED IN VERTICAL FLIGHT FOR VARIOUS PAYLOAD VOLUMES ASSUMING CONSTANT FUEL AND PAYLOAD WEIGHT FOR ALL CONDITIONS (FIGURE 10).

THE PAYLOAD VOLUME WAS ASSUMED TO BE INCREASED BY ADDING VARIOUS LENGTHS TO THE CYLINDRICAL SECTION OF THE BODY FORWARD OF THE FUEL TANKS, AND THE PAYLOAD WEIGHT WAS CONSIDERED TO BE EVENLY DISTRIBUTED OVER THIS VOLUME.

SINCE THE DESIRED MAXIMUM ALTITUDE WAS 500,000 FEET OR MORE FOR THE BASIC CONFIGURATION, FIGURE 10 SHOWS A MAXIMUM PAYLOAD VOLUME OF 43 CUBIC FEET USING A 300-POUND PAYLOAD WITH NO INCREASE IN FUEL TO ACHIEVE THIS OBJECTIVE. GREATER PAYLOAD VOLUMES MAY BE USED AND STILL ATTAIN AN ALTITUDE OF 500,000 FEET IF THE PAYLOAD WEIGHT IS REDUCED BY AN AMOUNT EQUIVALENT TO THE INCREASE IN STRUCTURAL WEIGHT. THE MINIMUM PAYLOAD WEIGHT CONSIDERED ON THE ABOVE CONDITIONS MUST BE GOVERNED BY THE STABILITY LIMITS PRESENTED IN FIGURE 9.

2. AN ADDITIONAL INVESTIGATION REQUESTED BY THE USAF WAS A STUDY OF THE AMOUNT OF ADDITIONAL FUEL NECESSARY TO OBTAIN A MILLION FEET ALTITUDE IN VERTICAL FLIGHT WITH A MODIFIED TEST VEHICLE CARRYING A 100-POUND PAYLOAD.

TO OBTAIN A MILLION FEET ALTITUDE, IT IS NECESSARY TO INCREASE THE FUEL WEIGHT FROM 2915 POUNDS TO 4260 POUNDS WHICH MAKES IT NECESSARY TO INCREASE THE BODY LENGTH FROM 362 INCHES TO 416 INCHES. THE FOLLOWING COMPARATIVE PERFORMANCE DATA ARE GIVEN FOR THE PRESENT BASIC CONFIGURATION AND THE PROPOSED CONFIGURATION.

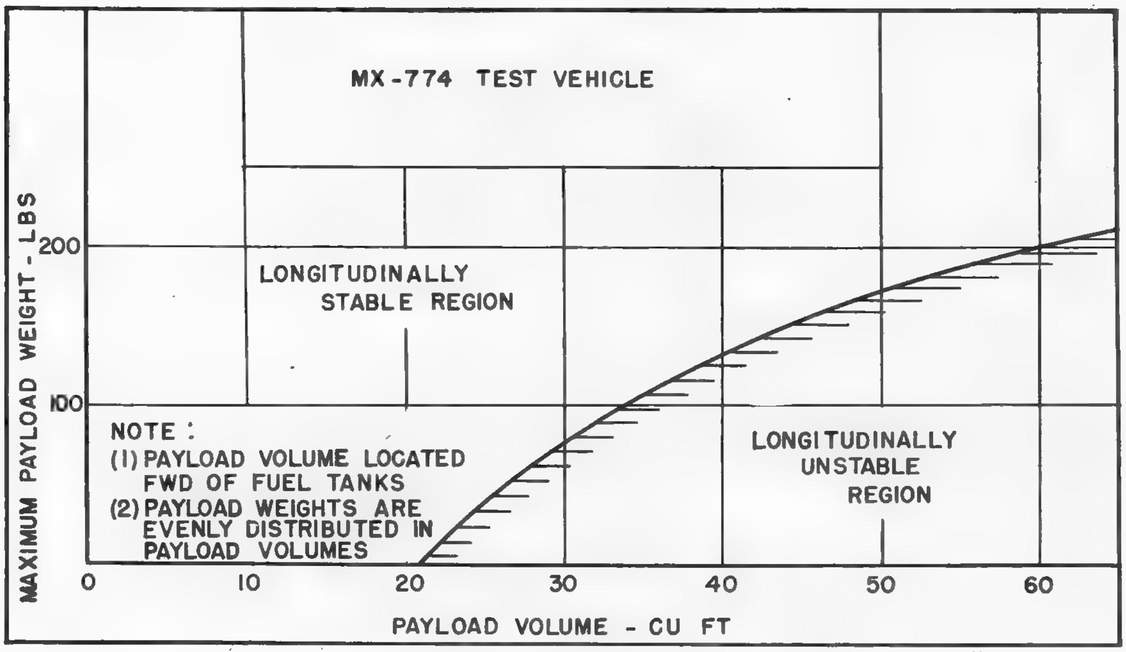


FIG 9 -- MINIMUM PERMISSIBLE PAYLOAD WEIGHT FOR STABILITY

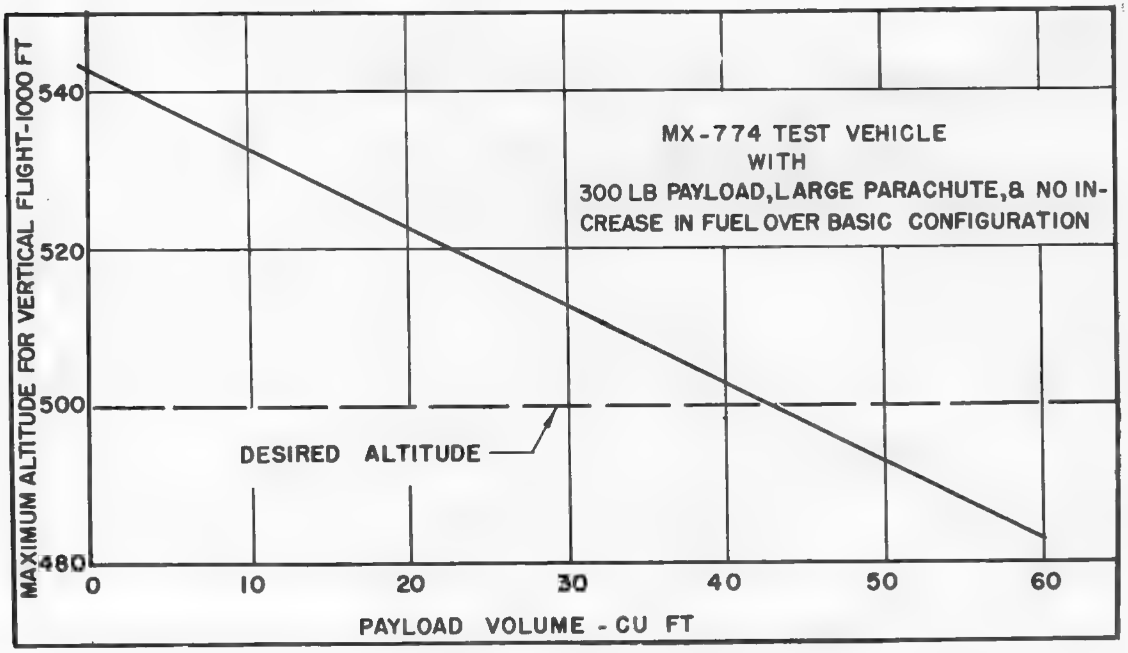


FIG 10 -- MAXIMUM ALTITUDE FOR VERTICAL FLIGHT

	PRESENT BASIC CONFIGURATION		PROPOSED CONFIGURATION	
GROSS WEIGHT	4,226	LB	5,400	LB
WEIGHT EMPTY	1,311	LB	1,140	LB
PAYLOAD WEIGHT	300	LB	100	LB
TYPE OF PARACHUTE	LARGE		NONE	
BODY LENGTH	362	IN.	416	IN.
FIN AREA (PER PANEL)	10.02	SQ FT	10.02	SQ FT
MAX BODY DIAMETER	30	IN.	30	IN.
THRUST AT SEA LEVEL	8,000	LB	8,000	LB
INITIAL ACCELERATION	1.89		1.48	
TIME OF POWERED FLIGHT	76.59	SEC	112.1	SEC
VELOCITY AT END OF BURNING	5,051	FT/SEC	7,238	FT/SEC
ALTITUDE AT END OF BURNING	130,000	FT	230,000	FT
MAXIMUM ALTITUDE	521,000	FT	1,042,000	FT

3. THE FINAL AERODYNAMIC REPORT ON THE BASIC CONFIGURATION OF THE TEST VEHICLE IS NEARING COMPLETION.

B. AZIMUTH GUIDANCE

RESULTS OF THE FIRST MANUAL CALCULATION OF THE PERFORMANCE OF THE MISSILE WHEN GUIDED IN AZIMUTH WERE COMPLETED. DUE TO THE MANY MATHEMATICAL OPERATIONS INVOLVED AND ASSUMPTIONS NECESSARY HOWEVER, IT SEEMED ADVISABLE TO UTILIZE OTHER MEANS OF OBTAINING A SOLUTION. HENCE A TRANSFER FUNCTION ANALYSIS WAS MADE FOR SUBMISSION TO THE ANALYSIS LABORATORY AT CALIFORNIA INSTITUTE OF TECHNOLOGY FOR DETERMINATION OF OPTIMUM SERVO CONSTANTS. BECAUSE OF THE MISFIRE AT THE POINT LOMA STATIC TEST HOWEVER, IT HAS BEEN DECIDED TO DISCONTINUE THE GUIDANCE ANALYSIS FOR LACK OF FUNDS. THE ANALYSIS PREPARED WILL BE MODIFIED FOR PRESENTATION TO THE ANALOG COMPUTOR AS A YAW STABILIZATION PROBLEM.

C. TEST VEHICLE - CONTROLLED FLIGHT PATH IN PITCH

THE STEP BY STEP INTEGRATION OF THE FLIGHT PATH OF THE FIRST TEST VEHICLE IN PITCH MENTIONED IN REFERENCE 2 HAS BEEN ABANDONED. THIS METHOD BECAME TOO COMPLEX FOR THE ACCURACY REQUIRED, HENCE A MUCH SIMPLER APPROXIMATE METHOD WAS SUBSTITUTED. PRELIMINARY CALCULATIONS SHOWED THAT THE PITCH PROGRAMING WAS INADEQUATE FOR THE DESIRED RESULTS, SO IT WAS DECIDED TO UTILIZE AERODYNAMIC TABS IN PITCH TO SUPPLEMENT JET DEFLECTION AS WELL AS TO START PITCHING EARLIER.

THE MISSILE IS ASSUMED TO FLY VERTICALLY UNTIL A MACH NUMBER OF 1.8 IS REACHED, AT WHICH TIME THE PITCH JETS ARE FULLY DEFLECTED AND HELD UNTIL A GIVEN FLIGHT PATH ANGLE IS REACHED. THE JETS ARE THEN DEFLECTED IN SUCH A MANNER AS TO HOLD THIS FLIGHT PATH ANGLE CONSTANT UNTIL THE END OF BURNING.

IT WAS DECIDED TO CAUSE THE MISSILE TO REACH THE NECESSARY FLIGHT PATH ANGLE SEVERAL SECONDS BEFORE CUTOFF FOR SEVERAL REASONS. SINCE FUEL BURNING TIME IS ONLY APPROXIMATELY KNOWN, THE POSSIBILITY WOULD EXIST THAT THE FUEL WOULD BE EXHAUSTED BEFORE γ REACHED ITS REQUIRED MAGNITUDE. FURTHER, TIME MUST BE ALLOWED FOR OSCILLATIONS CAUSED BY A SUDDEN CHANGE IN JET DEFLECTION TO DAMP OUT AND FOR THE GUIDANCE SYSTEM TO STABILIZE.

IT WAS ASSUMED THAT THE VELOCITY ALONG THE FLIGHT PATH, MACH NUMBER, AND DYNAMIC PRESSURE, q, WERE THE SAME AS FOR VERTICAL FLIGHT THROUGHOUT THE REGION OF POWERED FLIGHT. DAMPING FORCES AND THE COMPONENT OF THRUST PERPENDICULAR TO THE FLIGHT PATH DUE TO ANGLE OF ATTACK WERE NEGLECTED. BEYOND FUEL SHUT-OFF DRAG WAS ALSO NEGLECTED. IT WAS FURTHER ASSUMED THAT THE MISSILE WAS KEPT IN TRIM AT ALL TIMES.

WITH THESE CONSIDERATIONS IN MIND, THE PERTINENT EQUATIONS OF MOTION, WRITTEN WITH RESPECT TO FLIGHT PATH AXES, BECOME

$$(T\ell_1 / 2) \sin \delta - N\ell_2 + M_T = 0$$

$$mV\dot{\gamma} = -(T/2) \sin \delta + W \sin \gamma + N - N_T$$

WHERE

T = TOTAL THRUST IN LB

ℓ_1 = DISTANCE FROM CG TO JET PIVOT POINT IN FT

N = NORMAL FORCE DUE TO ANGLE OF ATTACK IN LB

ℓ_2 = DISTANCE FROM CG TO CP IN FT

M_T = PITCHING MOMENT DUE TO TAB DEFLECTION IN FT-LB

m = MASS OF THE MISSILE IN SLUGS

V = VELOCITY ALONG THE FLIGHT PATH IN FT/SEC

W = WEIGHT OF THE MISSILE IN LB

N_T = NORMAL FORCE DUE TO TAB DEFLECTION IN LB

δ = JET DEFLECTION IN RADIANS

γ = FLIGHT PATH ANGLE MEASURED TO VERTICAL IN RADIANS

$\dot{\gamma} = d\gamma/dt$ IN RAD/SEC

SIX CASES WERE SELECTED FOR CONSIDERATION, AND IT WAS ESTIMATED THAT THE DESIRED END RESULTS WOULD FALL SOMEWHERE WITHIN THEIR RANGE.

(1) THRUST AT SEA LEVEL = 8000 POUNDS; SPECIFIC IMPULSE AT SEA LEVEL = 210 SECONDS; γ_F = 10 DEGREES; BURNING TIME = 76.0 SECONDS

(2) THRUST AT SEA LEVEL = 8000 POUNDS; SPECIFIC IMPULSE AT SEA LEVEL = 210 SECONDS; γ_F = 8 DEGREES; BURNING TIME = 76.0 SECONDS

(3) THRUST AT SEA LEVEL = 8000 POUNDS; SPECIFIC IMPULSE AT SEA LEVEL = 210 SECONDS; γ_F = 10 DEGREES; BURNING TIME = 72.0 SECONDS

(4) THRUST AT SEA LEVEL = 8000 POUNDS; SPECIFIC IMPULSE AT SEA LEVEL = 210 SECONDS; γ_F = 8 DEGREES; BURNING TIME = 72.0 SECONDS

(5) THRUST AT SEA LEVEL = 8800 POUNDS; SPECIFIC IMPULSE AT SEA LEVEL = 217 SECONDS; γ_F = 10 DEGREES; BURNING TIME = 72.0 SECONDS

(6) THRUST AT SEA LEVEL = 8800 POUNDS; SPECIFIC IMPULSE AT SEA LEVEL = 217 SECONDS; γ_F = 8 DEGREES; BURNING TIME = 72.0 SECONDS

WHERE γ_F IS THE FLIGHT PATH ANGLE AT FUEL SHUT-OFF.

A STEP BY STEP INTEGRATION WAS CARRIED OUT TO DETERMINE THE LENGTH OF TIME REQUIRED FOR THE MISSILE TO REACH FLIGHT PATH ANGLES OF 8 DEGREES AND 10 DEGREES. THIS WAS FOUND TO VARY BETWEEN 15 AND 20 SECONDS DEPENDING UPON THE CASE UNDER CONSIDERATION, BUT IN ALL CASES SEVERAL SECONDS REMAIN AFTER REACHING THE REQUIRED FLIGHT PATH ANGLE BEFORE FUEL CUTOFF. CASES 1 - 4 ARE PLOTTED IN FIGURES 11A AND 11B, SHOWING A RANGE VARIATION OF FROM 29.7 TO 54.8 MILES, CASES 5 AND 6 ARE PLOTTED IN FIGURE 11C AND SHOW A RANGE VARIATION OF FROM 50.2 TO 62.2 MILES.

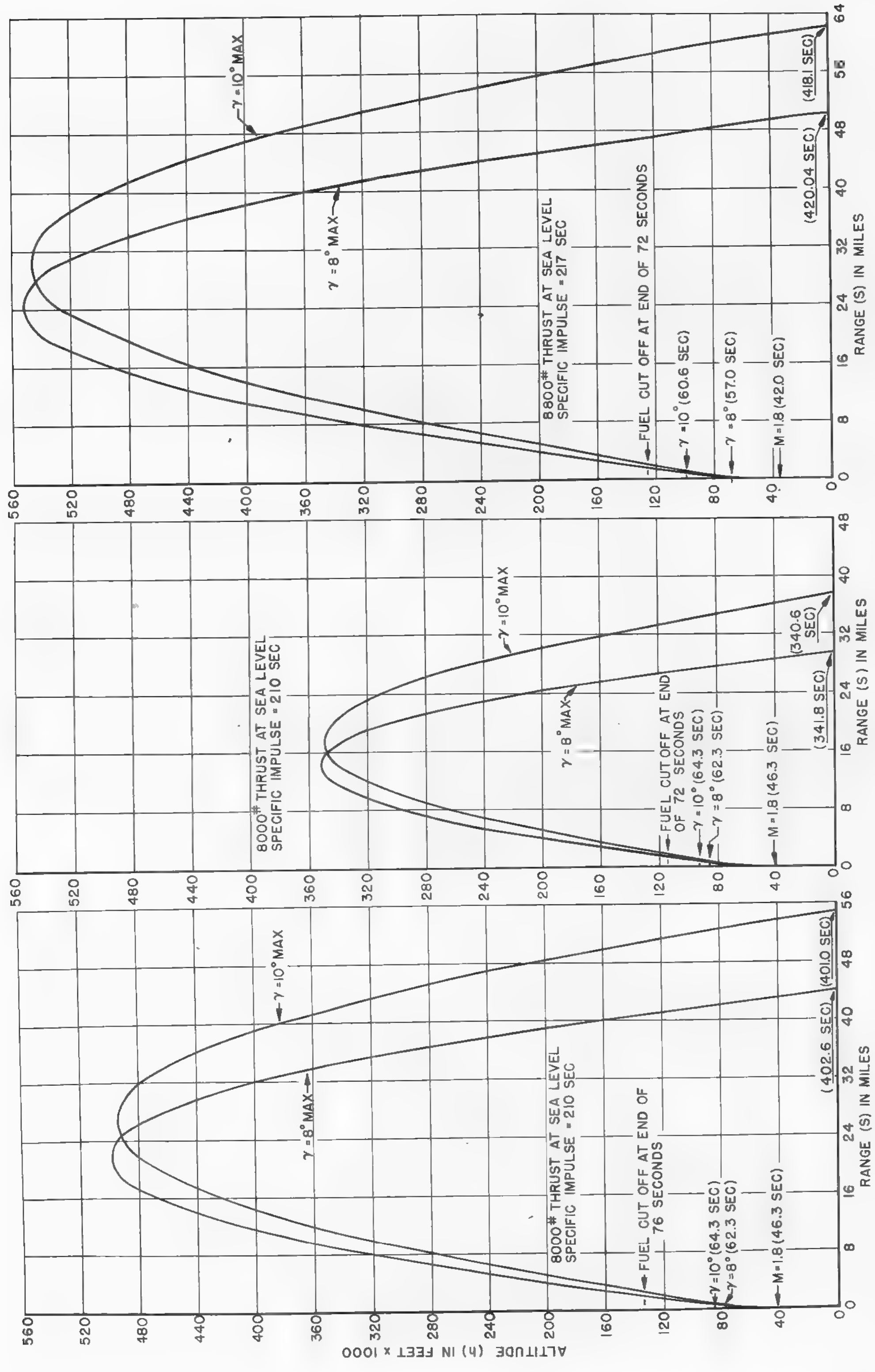


FIG 11 -- RANGE VERSUS ALTITUDE FOR SINGLE STAGE TEST VEHICLE

IN ORDER TO CAUSE THE MISSILE TO FOLLOW THE FLIGHT PATH DESIRED, THE REFERENCE ANGLE (θ_{REF}) OF THE PITCH GYRO MUST BE SHIFTED ACCORDING TO A PRE-CALCULATED TIME PLAN. THIS PROGRAM WAS CALCULATED FOR 8000-POUND THRUST AT SEA LEVEL FROM THE SAME EQUATIONS OF MOTION ASSUMED BEFORE AND RESULTS PLOTTED IN FIGURE 12. THE 25-DEGREE ANGLE QUOTED FOR THE PERIOD DURING WHICH THE FLIGHT PATH ANGLE IS CHANGING FROM VERTICAL TO EITHER 8 DEGREES OR 10 DEGREES FROM VERTICAL IS NOT CRITICAL, BUT MUST BE LARGE ENOUGH TO CAUSE THE PITCH JETS TO ROTATE AGAINST THEIR STOPS AND REMAIN THERE UNTIL THE DESIRED γ IS REACHED.

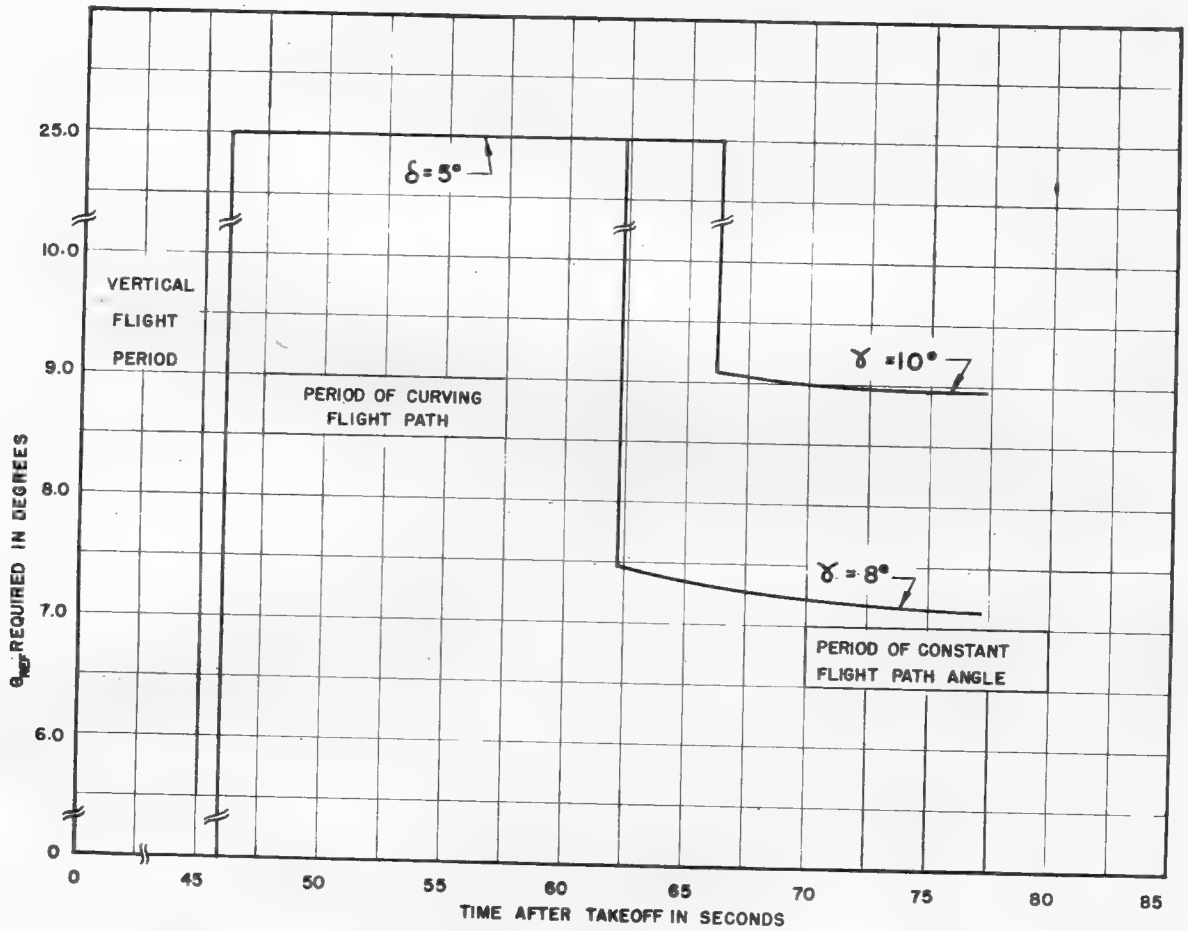


FIG 12 -- θ_{REF} REQUIRED FOR INDICATED FLIGHT PROGRAM

D. OPTIMUM SERVO CONSTANTS FOR STABILIZATION SYSTEM

THE TRANSFER FUNCTION ANALYSIS DESCRIBED IN REFERENCE 2, PAGES 24 TO 29, WAS SUBMITTED TO THE ANALYSIS LABORATORY AT CALIFORNIA INSTITUTE OF TECHNOLOGY FOR DETERMINATION OF OPTIMUM VALUES OF THE SERVO CONSTANTS ON THE ANALOG COMPUTOR.

VALUES OF K_ϕ, K_δ, AND K_F WHICH GAVE GOOD PERFORMANCE THROUGHOUT THE ENTIRE FLIGHT WERE FOUND TO BE

$K_\phi = 0.035$ VOLTS/DEGREE
$K_\delta = 0.53$ VOLTS/DEGREE
$K_F = 0.01$ VOLTS/DEGREE

THE OTHER SERVO CONSTANTS HAD BEEN DETERMINED PREVIOUSLY BY LABORATORY TESTS. AIRCRAFT RESPONSE CONSTANTS WERE DETERMINED AS A FUNCTION OF THE TIME OF FLIGHT FROM AERODYNAMIC DATA. THESE OPTIMUM CONSTANTS HAVE NOT YET BEEN TRIED IN THE SIMULATOR DUE TO PRACTICAL DIFFICULTIES ENCOUNTERED AND A DISCREPANCY BETWEEN THE SIMULATOR RESPONSE AND THE RESPONSE INDICATED BY THE TRANSFER FUNCTION ANALYSIS.

THE SET OF SERVO CONSTANTS WHICH HAS BEEN USED ON THE SIMULATOR IS:

$K_\phi = 0.13$ VOLTS/DEGREE (FREE GYRO)
$K_\delta = 0.07$ VOLTS/DEGREE (FEEDBACK)
$K_F = 0.07$ VOLTS/DEGREE (RATE GYRO)

THESE CONSTANTS GIVE STABLE ROLL CONTROL ON THE SIMULATOR. WHEN THESE CONSTANTS ARE USED IN EITHER THE TRANSFER FUNCTION ANALYSIS OR IN THE ANALOG COMPUTER THEY CAUSE THE SYSTEM TO BE UNSTABLE. THE REASON FOR THIS DISCREPANCY HAS NOT BEEN DETERMINED BUT SEVERAL POSSIBILITIES EXIST,

(1) IN BOTH THE TRANSFER-FUNCTION ANALYSIS AND ANALOG-COMPUTER STUDY, LINEAR AMPLIFIER-HYDRAULIC-SYSTEM OPERATION WAS ASSUMED. THE TIME LAG AND ATTENUATION IN THE ACTUAL SYSTEM MAY BE CONSIDERABLY LESS THAN THAT OF A LINEAR SYSTEM.

(2) THE STABILIZING EFFECT OF VISCOUS AND STATIC FRICTION PRESENT IN THE SIMULATOR TABLE IN ITS ROLL MOTION MAY BE APPRECIABLE. THE COMBINATION OF THE TWO MAY HAVE A GREATER STABILIZING EFFECT THAN ANY AERODYNAMIC DAMPING CONSIDERED IN THE TWO THEORETICAL CASES.

(3) THE SIMULATOR DOES NOT SIMULATE THE ACTUAL MISSILE.

(4) THE SERVO CONSTANTS MAY HAVE BEEN DETERMINED INCORRECTLY.

FURTHER LABORATORY TESTS WILL INDICATE THE REASON FOR THE DISCREPANCY.

E. INVESTIGATION OF LONG RANGE TRAJECTORIES

DURING THE SEPTEMBER, OCTOBER, AND NOVEMBER REPORT PERIOD, ATTEMPTS WERE MADE TO DEVELOP SIMPLER WAYS OF OBTAINING THE SPACE RANGE BETWEEN LAUNCHING SITE AND TARGET ON THE ROTATING EARTH. THE MOST SUCCESSFUL METHOD OF COMPUTATION APPEARS TO BE THE USE OF SUCCESSIVE APPROXIMATIONS. HOWEVER, MEANS HAVE BEEN DEVELOPED FOR EFFECTING A CONSIDERABLE REDUCTION IN THE CALCULATIONS REQUIRED FOR A SERIES OF APPROXIMATIONS.

THESE COMPUTATIONS FOR THE ROTATING EARTH HAVE BEEN FOR EASTWARD DIRECTIONS ALONG THE EQUATOR. FOR WESTWARD FLIGHTS ALONG THE EQUATOR, HARDLY MORE THAN A CHANGE OF SIGN IS INVOLVED FOR SIMILAR COMPUTATIONS. FOR OBLIQUE RANGES IN THE LATITUDES, A RELATIVELY SMALL AMOUNT OF ADDITIONAL COMPUTATION IS REQUIRED, IN WHICH THE SIMPLE COSINES OF LATITUDE FUNCTIONS ARE READILY INTRODUCED AND ROUTINE GREAT CIRCLE DISTANCES TO THE TARGET, AS AFFECTED BY EARTH ROTATION DURING TIME-IN-FLIGHT, ARE REQUIRED IN ACCORDANCE WITH STANDARD NAVIGATIONAL PRACTICE.

DATA FOR A RANGE OF 4500 MILES BETWEEN FUEL SHUT-OFF POINT AND TARGET ARE BEING CALCULATED. SHUT-OFF IS PRESUMED TO OCCUR 500 MILES FROM THE LAUNCHING SITE AND AT A HEIGHT OF NOT OVER 200 MILES. WHEN COMPLETED, THESE DATA MAY BE CONVERTED INTO GRAPHS SHOWING SHUT-OFF VELOCITIES AND TIME-IN-FLIGHT FOR 4500 MILE TRAJECTORIES WHICH START WITH A VELOCITY-VECTOR ELEVATION ANGLE OF 15 DEGREES. IT IS PLANNED TO EXTEND THE CALCULATIONS TO OTHER ANGLES NEAR 15 DEGREES.

FOR PURPOSES OF COMPARISON, A MINIMUM ENERGY TRAJECTORY OF 4500 MILES FROM A FUEL SHUT-OFF POINT AT A HEIGHT EQUAL TO 0.05 PERCENT OF THE EARTH'S RADIUS (OR ABOUT 200 MILES ALTITUDE) HAS BEEN CALCULATED. THE MAGNITUDE AND ANGLE OF ELEVATION OF THE VELOCITY VECTOR AT SHUT-OFF WERE FOUND TO BE 20,701.05 FEET PER SECOND AND 27 DEGREES-38 MINUTES-29.0 SECONDS, RESPECTIVELY. THE TIME-IN-FLIGHT FOR THIS RANGE ON A NON-ROTATING EARTH WAS FOUND TO BE 1635.54 SECONDS.

F. DISSOCIATION OF BOUNDARY LAYER GASES

INVESTIGATION OF THE EFFECT OF HEAT ABSORBING DISSOCIATION REACTIONS IN REDUCING BOUNDARY LAYER TEMPERATURES AT VARIOUS SPEEDS AND AMBIENT PRESSURES HAS PROVIDED ADDITIONAL INFORMATION IN THE HYPERSONIC SPEED RANGE.

FIGURE 7D OF PAGE 35 OF REFERENCE 3 IS HEREIN EXTENDED OVER THE ENTIRE RANGE OF ALTITUDES SIGNIFICANT IN THE WARHEAD SKIN HEATING PROBLEM OF THE MISSILE OF PROJECT MX-774. CHEMICAL EQUILIBRIUM OF THE TYPES OF REACTIONS CONSIDERED IS ASSUMED COMPLETE AT ALL TIMES. OTHER REACTIONS, INVOLVING ACTIVATED STATES OF MOLECULES AND ATOMS AND ELECTRICALLY CHARGED PARTICLES, WERE NOT CONSIDERED BECAUSE NO DATA ARE AVAILABLE ON EQUILIBRIUM CONSTANTS AND HEATS OF REACTION ON WHICH TO BASE A QUANTITATIVE ANALYSIS. DISSOCIATION CONSTANTS WERE TAKEN FROM REFERENCE 4. MOLAR SPECIFIC HEATS WERE TAKEN FROM REFERENCE 5.

THE CHEMICAL EQUILIBRIUM EQUATIONS OF FORMATION OF NO, O, AND N ARE THE BASIS OF THE CALCULATION OF DISSOCIATION.

$$1/2\ N_2 + 1/2\ O_2 \rightleftarrows NO$$

$$O_2 \rightleftarrows 2O$$

$$N_2 \rightleftarrows 2N$$

LETTING

X = NUMBER OF MOLES OF NO THAT ARE FORMED

Y = NUMBER OF MOLES OF O FORMED

AND Z = NUMBER OF MOLES OF N FORMED

THEN THE EQUILIBRIUM EQUATIONS ARE:

$$K_{P_{NO}} = \frac{X}{\sqrt{\left[(N_2) - (X/2) - (Z/2)\right]\ \left[(O_2) - (X/2) - (Y/2)\right]}} \qquad \text{Eq (1)}$$

$$K_{P_O} = \frac{Y^2\ P}{\left[(O_2) - (X/2) - (Y/2)\right]\ \left[1 + (Z/2) + (Y/2)\right]} \qquad \text{Eq (2)}$$

$$K_{P_N} = \frac{Z^2 P}{\left[(N_2) - (X/2) - (Z/2)\right]\left[1 + (Z/2) + (Y/2)\right]} \qquad \text{Eq (3)}$$

WHERE P IS THE TOTAL PRESSURE IN ATMOSPHERES, THE EQUILIBRIUM CONSTANTS K ARE AS INDICATED BY SUBSCRIPTS, AND THE PARENTHETICAL QUANTITIES ARE MOLAR CONCENTRATIONS OF THE INDICATED SUBSTANCES.

THE EQUATION $NO \rightleftharpoons N + O$ IS REDUNDANT BECAUSE IT'S EQUILIBRIUM CONSTANT MAY BE EXPRESSED IN TERMS OF THE ABOVE EQUILIBRIUM CONSTANTS AS FOLLOWS:

$$NO \rightleftharpoons N + O$$

$$K_4 = \frac{(N)\ (O)}{(NO)}$$

$$= \frac{(N)}{\sqrt{(N_2)}} \times \frac{(O)}{\sqrt{(O_2)}} \times \frac{\sqrt{(N_2)\ (O_2)}}{(N\ O)}$$

$$= \frac{\sqrt{K_3\ K_2}}{K_1}$$

AIR WAS TAKEN AS O_2 = 21 PERCENT, N_2 = 78 PERCENT, AND A = 1 PERCENT BY VOLUME.

SOLVING OF THE THREE EQUILIBRIUM EQUATIONS SIMULTANEOUSLY IS NOT SIMPLE. A COMBINATION OF GRAPHICAL AND TRIAL-AND-ERROR METHODS WAS USED FOR THEIR SOLUTION. EQUATION (1) WAS PLOTTED GRAPHICALLY ON A SERIES OF CURVES OF $\log_{10} X$ VERSUS $1/T^{o}$ KELVIN FOR VALUES OF Y RANGING FROM 0 TO 0.4 AND VALUES OF Z FROM 0 TO 1.4, THE RANGE OF $\log_{10} X$ BEING FROM -0.6 TO -2.0 AND 1/T RANGING FROM .00018 TO .00050. VERY SMALL INACCURACIES OCCURRED IN THE USE OF THESE GRAPHS OUTSIDE THESE RANGES, BUT THE ERROR INVOLVED IN THE FINAL RESULT IS BELIEVED TO BE NEGLIGIBLE.

BY USING THESE GRAPHS, THE TIME FOR THE TRIAL-AND-ERROR SOLUTION OF THE THREE SIMULTANEOUS EQUILIBRIUM EQUATIONS WAS REDUCED MATERIALLY.

When values of X, Y, and Z were found for a range of temperatures and pressures, the required heat to achieve this condition from 0 degrees Centigrade was computed. The terms appearing in this heat summation are as follows:

Heat of NO formation = X times ΔE_{NO}

Heat of O formation = Y times ΔE_{O}

Heat of N formation = Z times ΔE_{N}

O_2 heating = $\left(0.21 - \frac{X + Y}{2}\right)$ times C_p (molar) ΔT

N_2 heating = $\left(0.78 - \frac{X + Z}{2}\right)$ times C_p (molar) ΔT

NO heating = X times C_p (molar) ΔT

Formation of Y / 2 moles = Y (RT / 2)

Formation of Z / 2 moles = Z (RT / 2)

O heating = 5 Y ΔT

N heating = 5 Z ΔT

Argon heating = .01 times 5 T

The units of this summation are calories per mole weight of original air. This amount was converted to BTU per pound of original air by the factor 1.8 / 28.97.

Since the boundary layer temperature was desired as a function of flight velocity, and assuming that the total energy of the free stream being brought to rest in the boundary layer is expressed as an increase in enthalpy, flight velocity was obtained by,

$$V = \sqrt{2gJ\,\Delta H} \text{ ft/sec} \qquad \text{Eq (4)}$$

where ΔH is the summation of heat above multiplied by 1.8 / 28.97.

Figure 13 is a plot of T versus v with ambient pressure as a third parameter. ΔH may be computed from Figure 13 with the aid of equation (4). The factor N is that discussed in Reference 3.

Two small errors are present in the calculation of the summation of heat. The dissociation reaction was assumed to take place at 0 degrees Centigrade. However, values of ΔE for 0 degrees Kelvin were used instead. Also, the value of T used in calculating the heat of formation of new moles of O and N was from 0 degrees Centigrade rather than from 0 degrees Kelvin. The magnitude of the first error is approximately one percent for the dissociation reaction, the value computed being high. However, the error on the summation is somewhat less than one percent due to the added heating terms. The error in the second case depends on the temperature and is a greater proportion at the lower temperatures, but because this term is so small in comparison with the rest of the heat, its effect on the final result is almost negligible. This error decreases the value of the summation of heat, so that the two errors exert a compensating effect on each other.

The curves of Figure 13 have been plotted from 3 points on each curve: 8541 degrees F (5000 degrees K), 5842 degrees F (3500 degrees K), and 3140 degrees F (2000 degrees K), with the exception of curves for P = 1, and 10 atmospheres where the lowest temperature is 4042 degrees F (2500 degrees K). As the calculation of more points would be a tedious process, it was deemed to be unnecessary at this time. However, when plotting, the curves were faired so as to take advantage of the position of points on adjacent curves, so that the accuracy may not be so far off at intermediate points as might at first be expected.

The data as calculated for Figure 13 being available, values of C_p and γ for air, including dissociation effects, were calculated and are presented as Figures 14 and 15.

The equations for these values are:

$$C_p = \frac{\Delta H}{\Delta T(N)} \quad \text{calories/mole at } T_2 \qquad \text{Eq (5)}$$

Where ΔH = the previously determined heat summation from 0 degrees Centigrade to T_2, $\Delta T = T_2$ degrees Centigrade, N = number of moles at T_2.

In order to be consistent with Figure 13, and to keep the more common engineering units, T_2 was plotted in degrees Kelvin.

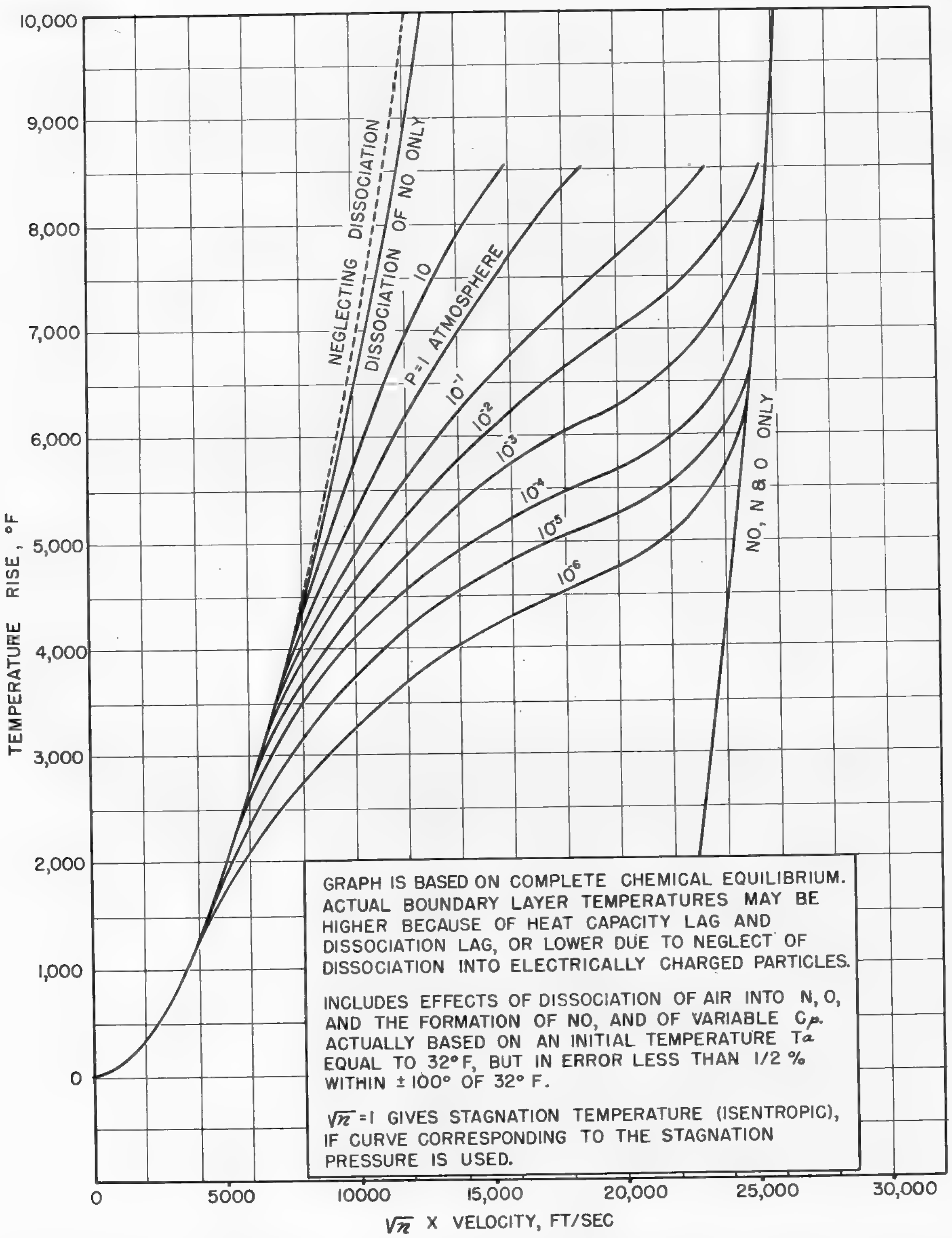


FIG 13 -- BOUNDARY LAYER TEMPERATURE RISE AS A FUNCTION OF VELOCITY AND BOUNDARY LAYER PRESSURE

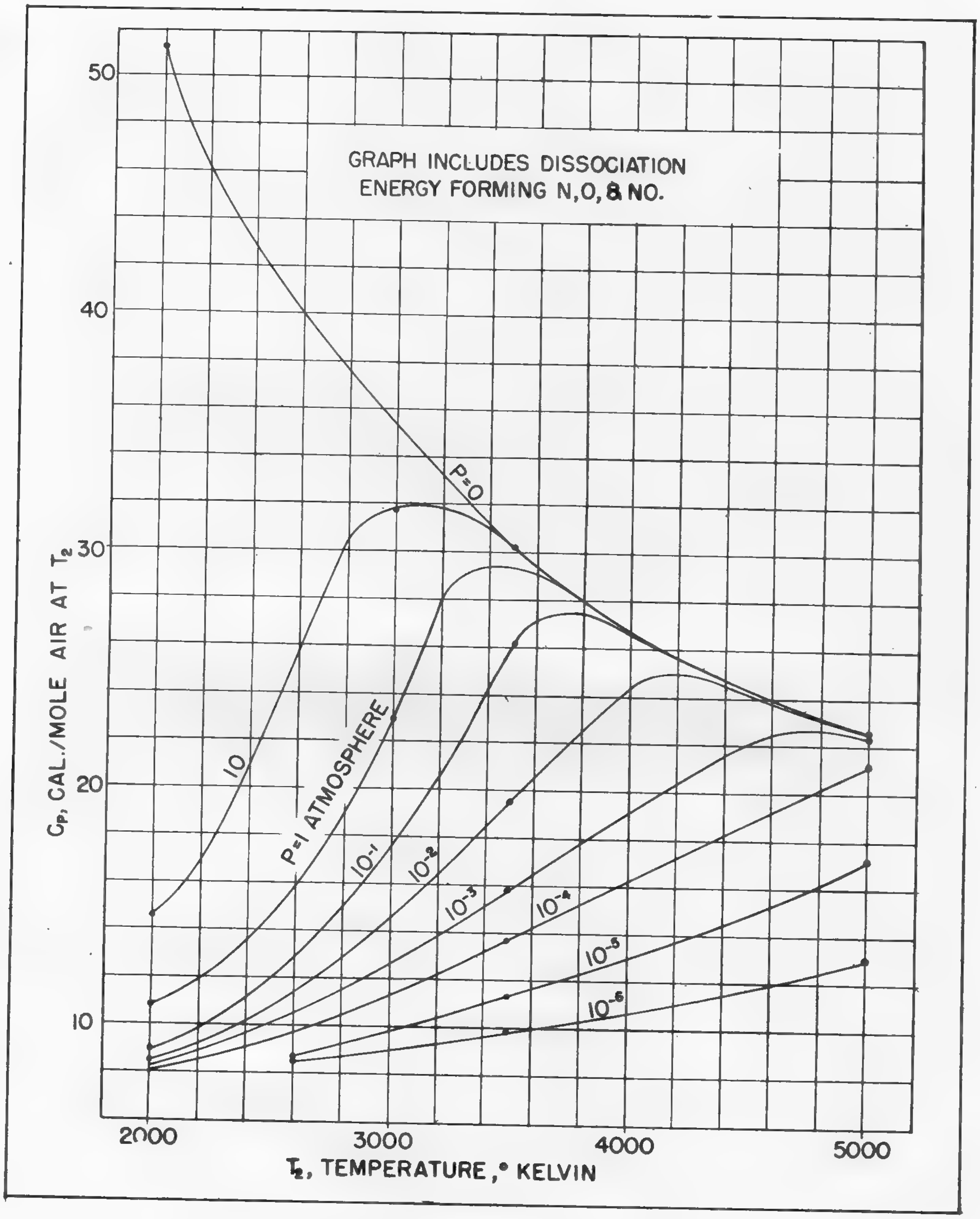


FIG 14 -- AVERAGE C_P FROM 0° TO T_2 FOR VARIOUS PRESSURE CONDITIONS

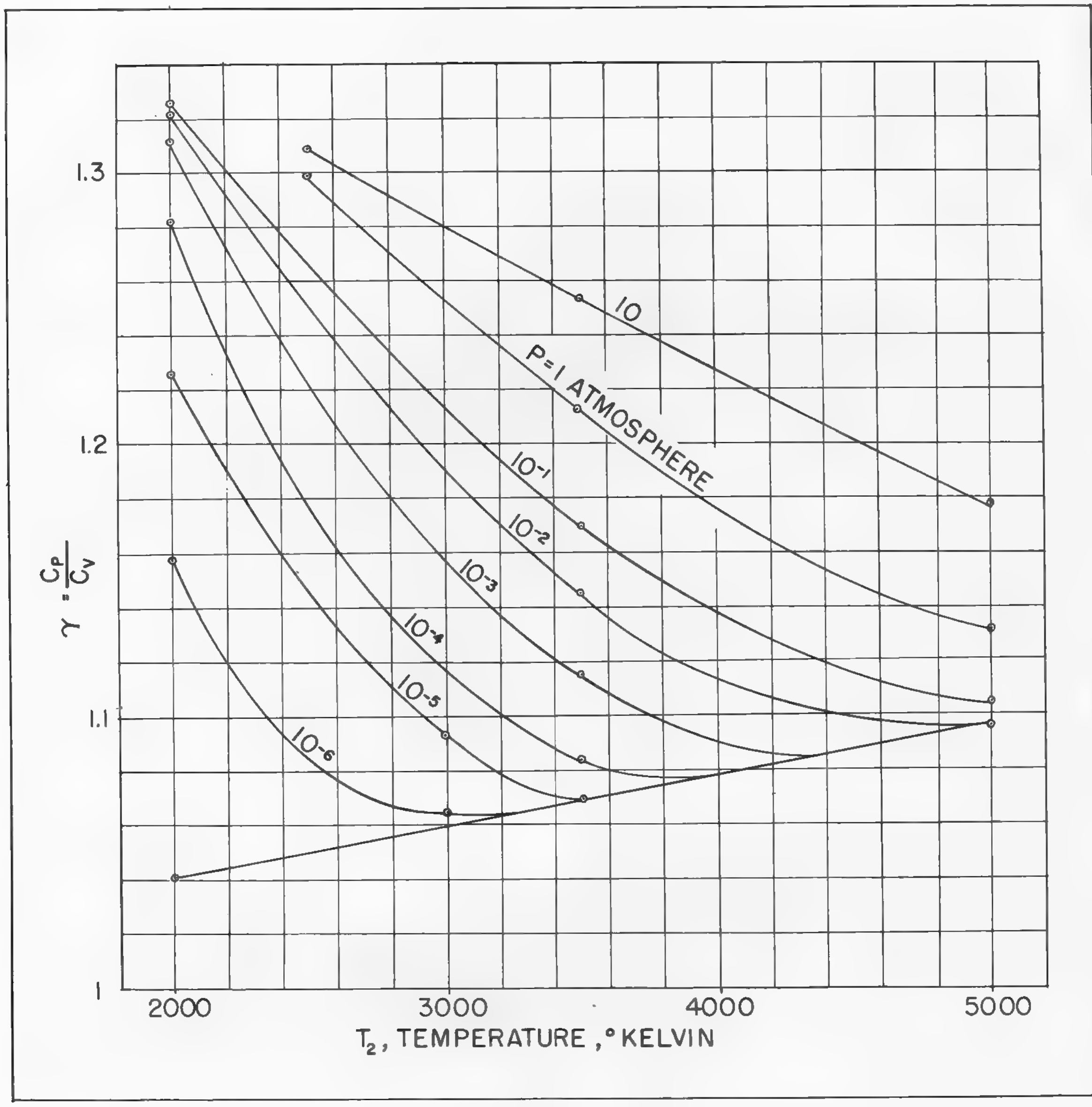


FIG 15 -- AVERAGE γ FROM 0° C TO T_2 FOR VARIOUS PRESSURE CONDITIONS

$$\gamma = 1 + \frac{R}{C_p - R}$$ WHERE R = UNIVERSAL GAS CONSTANT, 1.987 CALORIES/DEGREE KELVIN. C_p = MOLAR SPECIFIC HEAT AS ABOVE.

THE SAME DISCUSSION AS TO ERROR AND NUMBER OF COMPUTED POINTS FOR FIGURE 13 ALSO APPLIES TO FIGURES 14 AND 15.

G. EFFECT OF NOZZLE INSERTS ON ROCKET MOTOR PERFORMANCE

THE USE OF DROPPABLE INSERTS IN THE EXHAUST NOZZLE FOR THE ROCKET MOTOR UNDER CONSIDERATION FOR PROJECT MX-774 HAS BEEN FOUND TO INCREASE THE THRUST OF THE MOTOR APPRECIABLY AND UNDER CERTAIN CONDITIONS AS MUCH AS 10 PERCENT. CALCULATIONS WERE CARRIED OUT ON A MOTOR CONFIGURATION HAVING A ONE FOOT DIAMETER THROAT AND A SEVEN FOOT DIAMETER MAXIMUM EXIT SECTION SIMILAR TO THAT PROPOSED BY AEROJET, FIGURE 16.

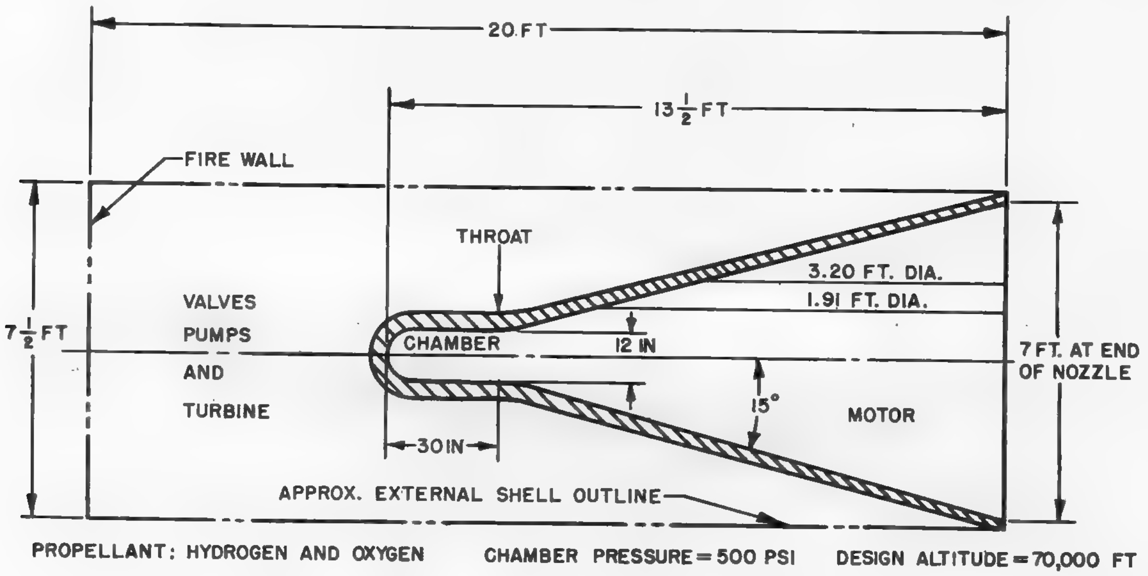


FIG 16 -- ROCKET MOTOR WITH VARIABLE NOZZLE

THE PROPELLANTS ASSUMED IN THE ANALYSIS WERE OXYGEN AND HYDROGEN USING A 100 PERCENT EXCESS OF HYDROGEN, THE BASIC EQUATION OF THE CHEMICAL REACTION BEING:

$$2H_2 + 1/2\ O_2 \rightleftharpoons H_2O + H_2$$

EFFECTS OF DISSOCIATION OF H_2O INTO H_2, H, O, OH, AND O_2 WERE INCLUDED IN THE ANALYSIS. COMPLETE CHEMICAL EQUILIBRIUM WAS ASSUMED AT ALL TIMES. ACTUAL QUANTITIES CALCULATED MAY BE IN ERROR BY 5 TO 10 PERCENT, BUT THE COMPARISON BETWEEN THE RESULTS SHOULD BE FAIRLY ACCURATE AS THE SAME BASIC DATA, WHICH WERE REDUCED TO SMOOTH CURVES IN THE CASE OF THE DISSOCIATION, WERE USED THROUGHOUT THE ANALYSIS. NO ATTEMPT WAS MADE TO COMPUTE THE THRUST OF THE MOTOR WITHOUT INSERTS IN THE OVEREXPANDED (OR UNDER ATMOSPHERIC) CONDITION FOLLOWED BY A SHOCK WAVE OR SERIES OF SHOCK WAVES.

G. EFFECT OF NOZZLE INSERTS ON ROCKET MOTOR PERFORMANCE

THE USE OF DROPPABLE INSERTS IN THE EXHAUST NOZZLE FOR THE ROCKET MOTOR UNDER CONSIDERATION FOR PROJECT MX-774 HAS BEEN FOUND TO INCREASE THE THRUST OF THE MOTOR APPRECIABLY AND UNDER CERTAIN CONDITIONS AS MUCH AS 10 PERCENT. CALCULATIONS WERE CARRIED OUT ON A MOTOR CONFIGURATION HAVING A ONE FOOT DIAMETER THROAT AND A SEVEN FOOT DIAMETER MAXIMUM EXIT SECTION SIMILAR TO THAT PROPOSED BY AEROJET, FIGURE 16.

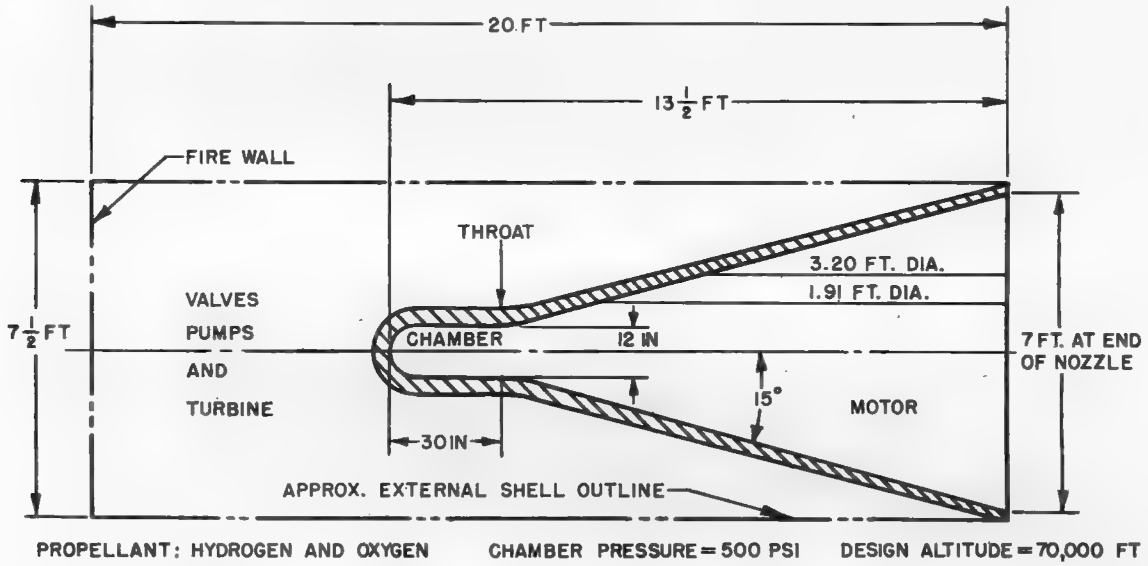


FIG 16 -- ROCKET MOTOR WITH VARIABLE NOZZLE

THE PROPELLANTS ASSUMED IN THE ANALYSIS WERE OXYGEN AND HYDROGEN USING A 100 PERCENT EXCESS OF HYDROGEN, THE BASIC EQUATION OF THE CHEMICAL REACTION BEING:

$$2H_2 + 1/2\ O_2 \rightleftharpoons H_2O + H_2$$

EFFECTS OF DISSOCIATION OF H_2O INTO H_2, H, O, OH, AND O_2 WERE INCLUDED IN THE ANALYSIS. COMPLETE CHEMICAL EQUILIBRIUM WAS ASSUMED AT ALL TIMES. ACTUAL QUANTITIES CALCULATED MAY BE IN ERROR BY 5 TO 10 PERCENT, BUT THE COMPARISON BETWEEN THE RESULTS SHOULD BE FAIRLY ACCURATE AS THE SAME BASIC DATA, WHICH WERE REDUCED TO SMOOTH CURVES IN THE CASE OF THE DISSOCIATION, WERE USED THROUGHOUT THE ANALYSIS. NO ATTEMPT WAS MADE TO COMPUTE THE THRUST OF THE MOTOR WITHOUT INSERTS IN THE OVEREXPANDED (OR UNDER ATMOSPHERIC) CONDITION FOLLOWED BY A SHOCK WAVE OR SERIES OF SHOCK WAVES.

CHEMICAL EQUILIBRIA WERE CALCULATED AT 300 PSIA AND VARIOUS TEMPERATURES UNTIL THE HEAT RELEASED BY THE REACTION EQUALED THE HEAT REQUIRED TO RAISE THE TEMPERATURE OF THE PRODUCTS TO THE ASSUMED TEMPERATURE; THESE ARE COMBUSTION CHAMBER STAGNATION CONDITIONS. AT A LOWER TEMPERATURE AND PRESSURE γ WAS COMPUTED AND THE TEMPERATURE RATIO OBTAINED FROM THE EQUATION

$$T_{CHAMBER} / T_{THROAT} = 1 + \left[(\gamma - 1) / 2\right] M^2$$

WHERE $M = 1$ FOR THE NOZZLE THROAT. BY TRIAL AND ERROR, VALUES OF γ, T_T AND P_T WERE FOUND TO BE MUTUALLY CONSISTENT AT $M = 1$, THUS ESTABLISHING THE CONDITIONS AT THE NOZZLE THROAT, CHEMICAL AND THERMAL EQUILIBRIUM BEING AGAIN ESTABLISHED AS FOR THE COMBUSTION CHAMBER. P_T WAS CALCULATED FROM THE ADIABATIC THEORY.

CHEMICAL AND THERMAL EQUILIBRIA WERE AGAIN ESTABLISHED BY TRIAL AND ERROR FOR VARIOUS EXIT PRESSURES AND THE STATE OF THE EXHAUST PRODUCTS FOUND. KNOWING THE ENERGY RELEASED AND THE SPECIFIC VOLUME OF THE EXHAUST PRODUCTS, THE VELOCITY AND THEN THE EXIT AREA WERE COMPUTED. WHEN THE EXIT AREA REACHED A VALUE OF 38.5 SQUARE FEET, CORRESPONDING TO A MAXIMUM NOZZLE EXIT DIAMETER OF 7 FEET, THE PROCESS WAS TERMINATED.

FIGURE 17 SHOWS THE DIFFERENCE BETWEEN THE ENERGY RELEASED BY COMBUSTION AND THE ENERGY REQUIRED TO HEAT THE COMBUSTION PRODUCTS TO THE TEMPERATURE PLOTTED AS ABSCISSA FOR SELECTED POINTS IN COMPLETE CHEMICAL EQUILIBRIUM. THIS ENERGY DIFFERENCE IS EQUAL TO THE KINETIC ENERGY OF THE EXHAUST GAS ($E = v^2 / 2G$). HENCE THE EXHAUST VELOCITY IS KNOWN AS A FUNCTION OF EXIT PRESSURE. THE APPARENT BUMP IN THE CURVE OF FIGURE 17 OF HEAT ABSORBED BY HEATING AND DISSOCIATION IS POSSIBLY DUE TO THE VARIATION IN SPECIFIC HEATS REACHING A MAXIMUM RATE OF INCREASE WITH TEMPERATURE. DISSOCIATION ENERGY OF FIRST H_2O AND SECOND H_2 AND O_2 AT THE HIGHER TEMPERATURES THEN TENDS TO INCREASE THE SLOPE AGAIN.

THRUST WAS COMPUTED FROM THE EQUATION:

$$F = Mv + A_E (P_E - P_0)$$

FOR THE VARIOUS CONDITIONS CONSIDERED AND PLOTTED AGAINST ALTITUDE RATHER THAN P_0 IN FIGURE 18.

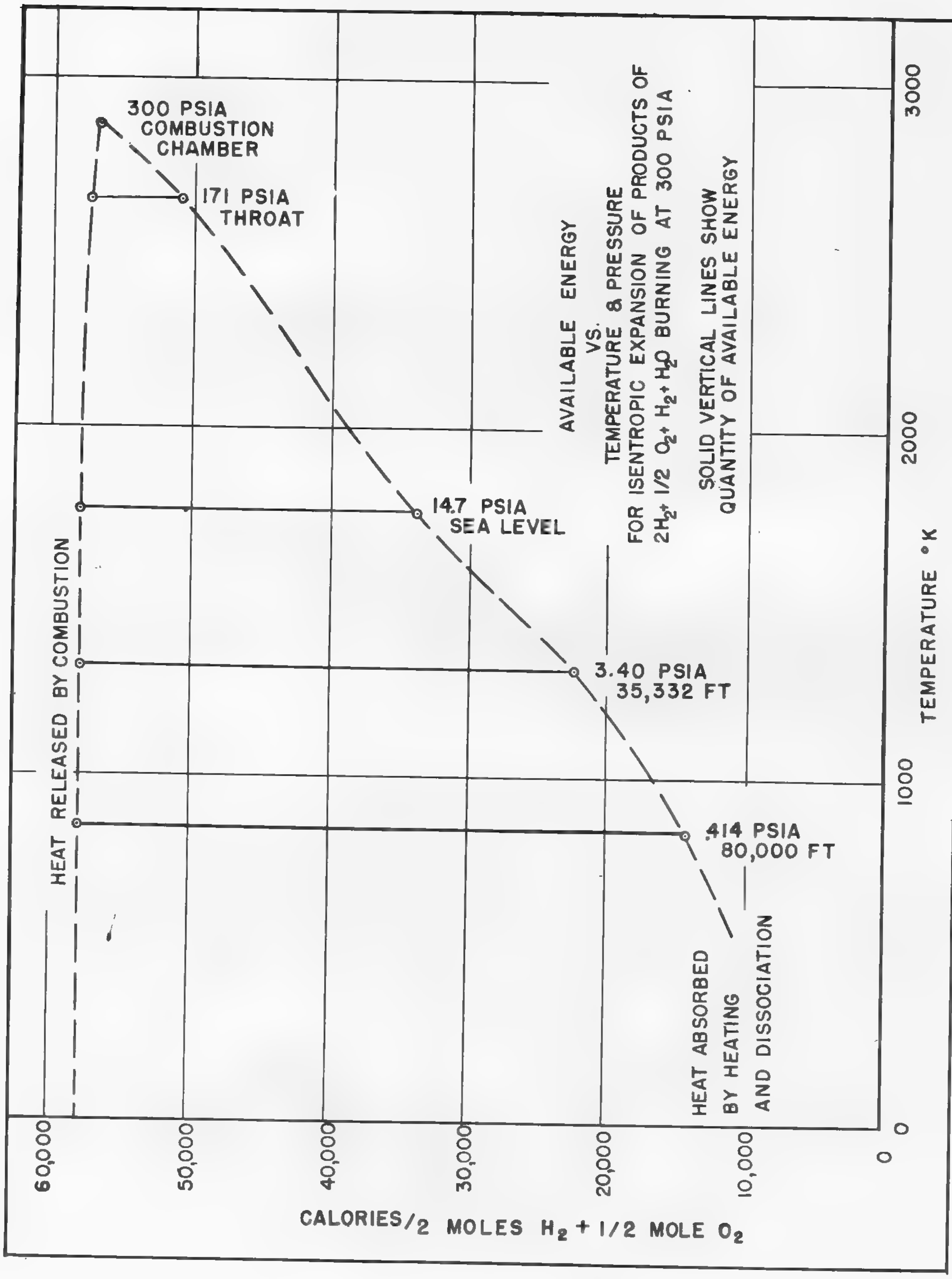


FIG 17 -- HEAT ENERGIES OF COMBUSTION AND ABSORPTION

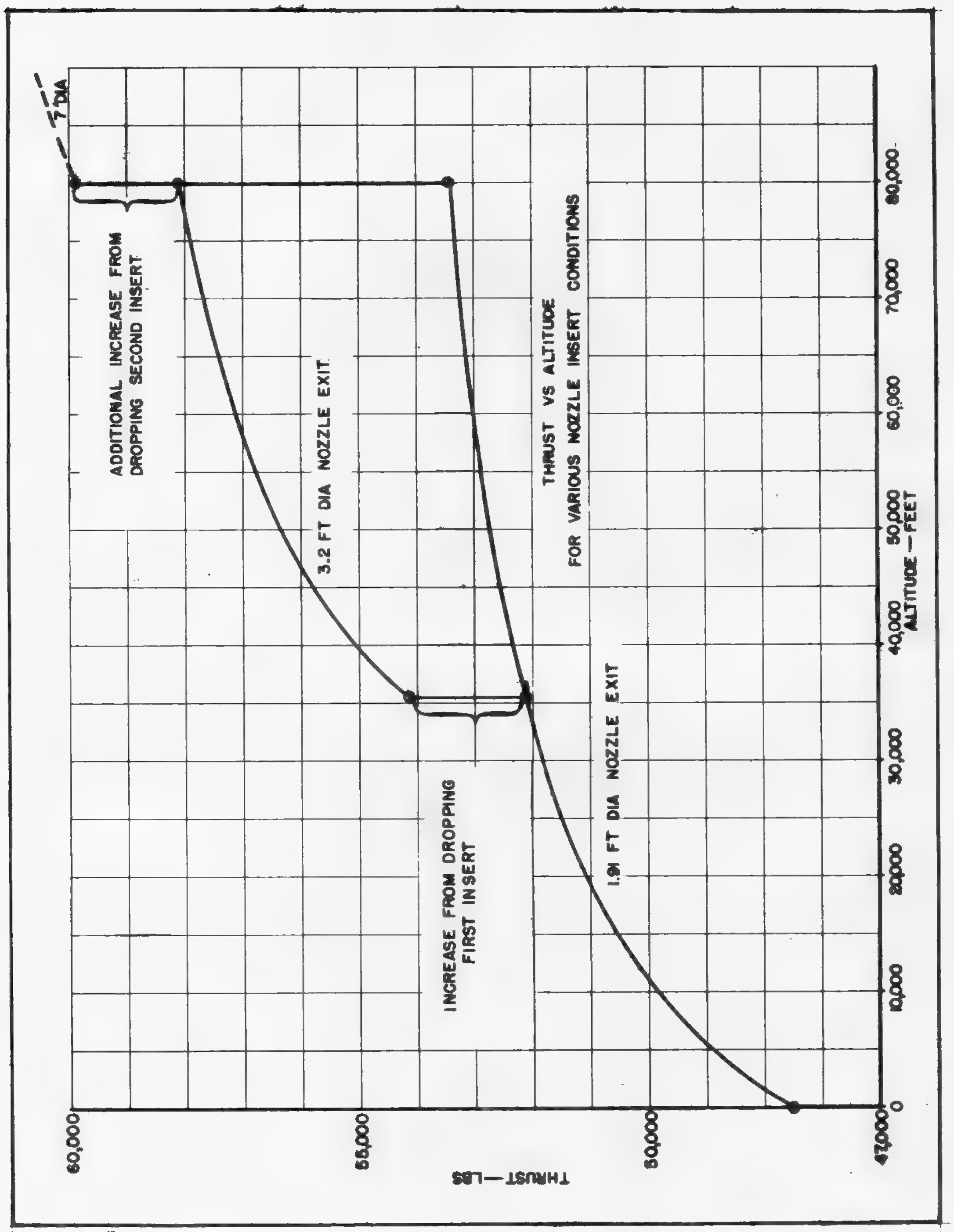


FIG 18 -- THRUST OF ROCKET HAVING CHAMBER INSERTS

SECTION V
PROPULSION AND FUEL

A. HYDROGEN PEROXIDE TANK

PRESSURE TEST OF THE H_2O_2 TANK PROVED IT TO BE UNDER-STRENGTH. THIS TANK IS OF WELDED ALUMINUM ALLOY CONSTRUCTION. WIRE WRAPPING WAS FOUND TO BE THE MOST SUITABLE METHOD OF RE-INFORCING THE TANK AND THE TANK WAS REDESIGNED. A WRAPPED TANK OF THE OLD DESIGN WITHSTOOD 425 PSI SO THE REDESIGN SHOULD END THE TROUBLE. IN THE INTERIM, A BREATHING-OXYGEN TANK WAS MOUNTED ON THE OUTER SURFACE OF THE STATIC FIRING TEST ARTICLE TO SERVE DURING STATIC TESTS AT POINT LOMA.

B. COLD CHECK AND STATIC FIRING TEST ON PROPULSION SYSTEM

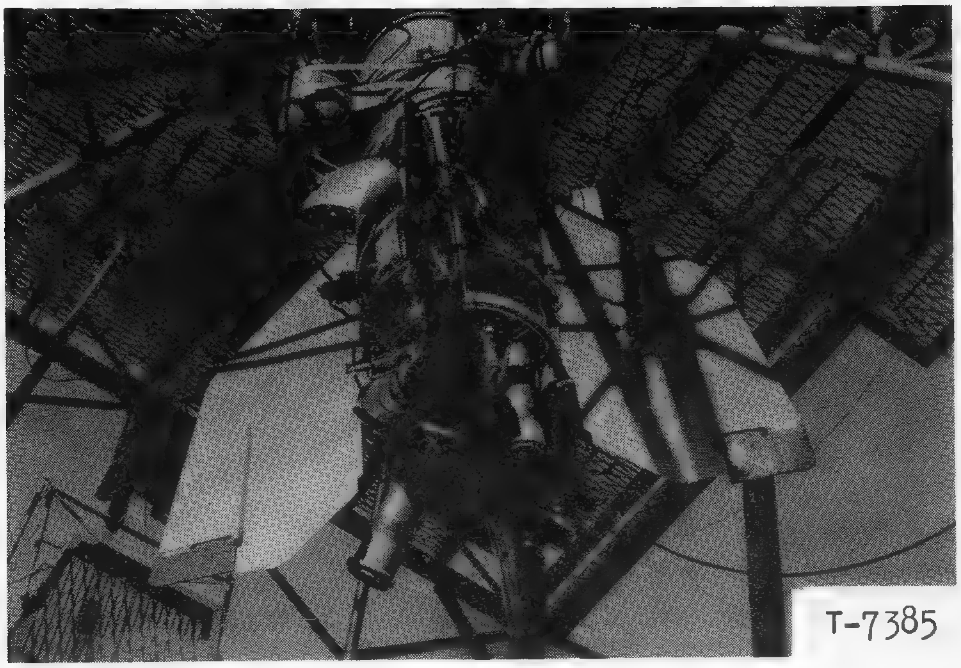
AN ACCOUNT OF THESE TESTS AT POINT LOMA IS GIVEN IN SECTION IX OF THIS REPORT.

C. FUTURE TEST OF POWER PLANT

THE DAMAGED ENGINE, FIGURE 19, IS TO BE REPAIRED IN CVAC'S SHOP. FOUR SPARE ENGINE CYLINDERS ARE TO BE ASSEMBLED ONTO THE MOUNT FOR RESUMPTION OF FIRING TESTS AT POINT LOMA. FOR THESE TESTS, ALL NON-ESSENTIAL MECHANICAL AND ELECTRONIC PARTS ARE TO BE LEFT OFF THE MISSILE. STEPS ARE BEING TAKEN TO PREVENT THE RECURRENCE OF FAULTY IGNITION.

TESTS AT POINT LOMA, WITH THE MISSILE CONTAINING ALL PROPOSED FLIGHT EQUIPMENT, ARE SCHEDULED TO BE RESUMED ABOUT 1 FEBRUARY 1948. THESE TESTS WILL BE CONDUCTED WITH EITHER THE REPAIRED ENGINE OR THE FIRST PRODUCTION ENGINE TO BE DE-LIVERED BY RMI.

ACCEPTANCE TESTS FOR THE FIRST PRODUCTION ENGINE WERE PROMISED BY RMI TO START AT THEIR PLANT ON 17 DECEMBER 1947. OUR PROPOSAL TO RUN ACCEPTANCE TESTS OF THE PRODUCTION ENGINE ON OUR STAND TOGETHER WITH ALL ENGINE EQUIPMENT UNDER NORMAL CONDITIONS OF USE WAS TURNED DOWN BY RMI.



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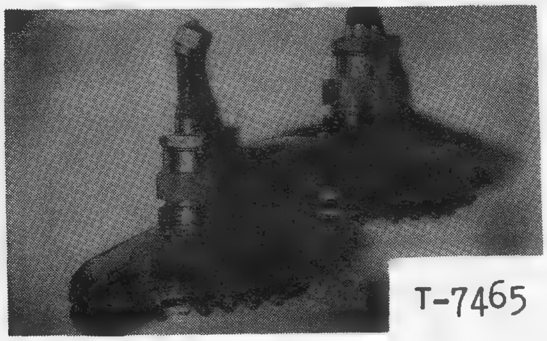

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FIG 19 -- DAMAGE TO ENGINE SECTION COMPONENTS DURING FIRST STATIC RUN

SECTION VI
STABILIZATION AND CONTROLS

A. EQUIPMENT INSTALLATION

Work on this phase consisted primarily of final laboratory testing of the amplifier and gyro panel assembly and installation of the system in the Static Test Missile. Preliminary operational tests, in preparation for the firing tests, were then conducted.

The No. 2 stabilization panel assembly, originally intended for Flight Missile No. 2 is now being completed for use in the No. 1 Flight Missile. This arrangement will permit more extensive use of the first panel in the ground test missile and will leave a complete system available for simulator tests during ground tests on the missile. The second panel is approximately 90 percent complete.

B. EQUIPMENT DEVELOPMENT

1. <u>Amplifiers</u>

With conclusion of tests performed under this program during the previous bi-monthly work period, no further work has been necessary on the amplifiers for the stabilization control system other than final checking and sealing before installation in the missile. Amplifiers have been constructed for two missile installations with an additional set to be used as a spare.

2. <u>Gyros</u>

Final operational testing and adjustments have been made on the gyros which are now installed in the Static Test Missile. A spare set of gyros for this missile has been completed and is ready for use.

Modifications found necessary as a result of tests on the first set of gyros have been made on the gyros for Flight Missile No. 1 and all spare sets. These modifications

INCLUDE CHANGE OF RATIOS ON THE AUTOSYN PICK-OFFS TO INCORPORATE NEW SERVO CONSTANTS, AS RECOMMENDED BY DYNAMICS STUDIES AND CONFIRMED BY SIMULATOR TESTS, AND ADDITION OF AUTOMATIC CAGING BUTTONS TO THE GYROS SO AS TO FACILITATE GROUND TESTS. WORK HAS BEEN DISCONTINUED ON THE PIONEER GYRO MODIFICATIONS AND LAYOUTS PROPOSED FOR INSTALLATION IN THE THIRD MISSILE PANEL ASSEMBLY.

3. SOLENOID CONTROL VALVES

THE ALL ALUMINUM CONTROL VALVE BLOCK ASSEMBLY, AS DESCRIBED IN REFERENCE 2, AND ONE SET OF INTEGRAL CONTROL AND SERVO RESPONSE UNITS HAVE BEEN GIVEN FINAL TESTS ON THE SIMULATOR. AFTER SATISFACTORY OPERATION THEY WERE INSTALLED IN THE STATIC TEST MISSILE. DURING THE FIRING TEST, THE VALVE ASSEMBLY WAS DAMAGED, REQUIRING IT TO BE REBUILT. A SECOND CONTROL VALVE FOR USE IN FLIGHT MISSILE NO. 1 HAS BEEN RECEIVED FROM THE VENDOR, AND IS CURRENTLY BEING ASSEMBLED WITH ITS CONTROL SOLENOIDS.

4. INTEGRAL CORRECTION AND RESPONSE UNIT

SOME DIFFICULTY WAS ENCOUNTERED IN USING THE ORIGINAL INTEGRAL CORRECTION AND RESPONSE UNITS. ROUGHNESS IN THE GEARING MADE IT NECESSARY TO OPERATE THE INTEGRAL SYSTEM AT A HIGHER GAIN SETTING THAN IS DESIRABLE. TO OVERCOME THIS TROUBLE, NEW MECHANISMS HAVE BEEN DESIGNED AND BUILT USING INSTRUMENT-TYPE GEARING. THESE UNITS HAVE BEEN TESTED WITH VERY SATISFACTORY RESULTS. FIGURE 20 SHOWS ONE OF THESE UNITS WITH THE COVERS REMOVED. THE EXTERNAL GEARING PROVIDES MEANS FOR HAND "ZEROING" AND DRIVES THE TELEMETERING POTENTIOMETER.

DURING THE FIRING TEST AT POINT LOMA, ONE SET OF THE INTEGRAL CORRECTION AND RESPONSE UNITS WAS DAMAGED SO AS TO REQUIRE REPLACEMENT. THE SET ORIGINALLY CONSTRUCTED FOR SPARES HAS BEEN INSTALLED IN THE STATIC TEST MISSILE AND ANOTHER SET OF SPARES IS READY FOR USE.

RESULTS OF THE INTEGRAL CORRECTION AND RESPONSE SYSTEM TEST CONDUCTED DURING THE SUBJECT REPORT PERIOD ARE SHOWN IN FIGURE 21. RATE OF INTEGRAL CORRECTION WAS FOUND TO BE PROPORTIONAL TO THE FREE GYRO ERROR SIGNAL OVER A RANGE OF PLUS OR MINUS 0.15 DEGREES. THE MAXIMUM RATE OF CORRECTION IS APPROXIMATELY 1.50 DEGREES PER SECOND.

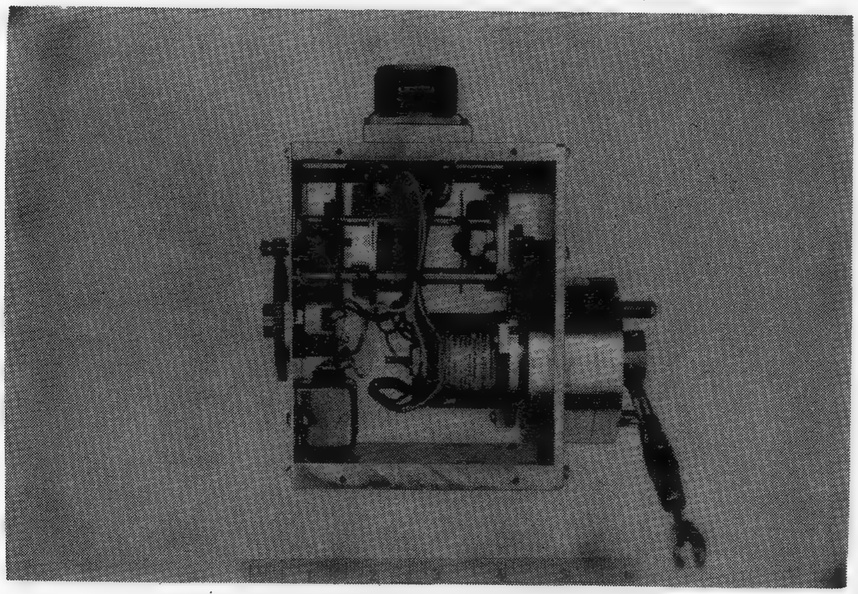

FIG 20 -- INTEGRAL CORRECTION AND RESPONSE UNIT MODEL 2

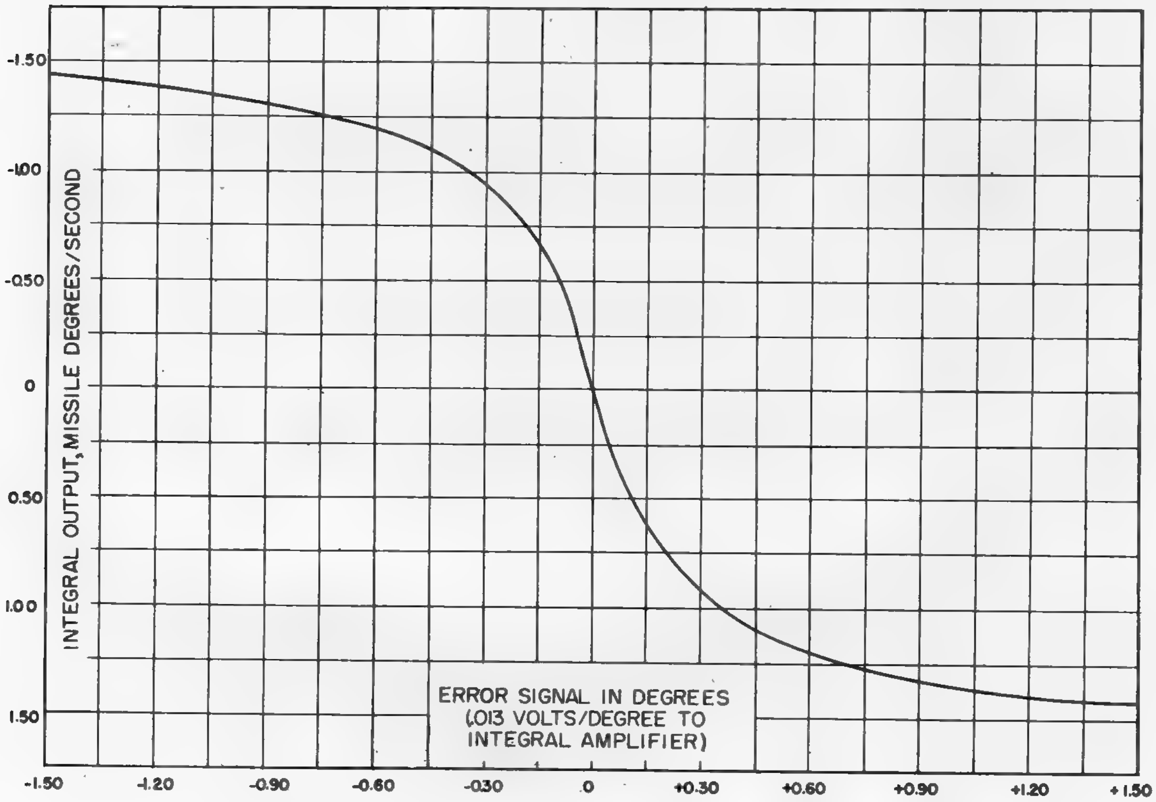

FIG 21 -- RESPONSE RATE VERSUS ERROR SIGNAL

5. SIMULATOR

FINAL TESTS ON THE STABILIZATION SYSTEM OF THE STATIC TEST MISSILE WERE RUN ON THE SIMULATOR WITH ARTIFICIAL LOADS IMPOSED BY METERING VALVES IN THE HYDRAULIC SYSTEM LINES. SERVO CYLINDER RESPONSE RATES WERE THEN COMPARED WITH RESULTS OF SEPARATE TESTS RUN ON A SPECIAL INERTIA AND SPRING LOAD TEST SET-UP. DYNAMIC LOAD TESTS AND SIMULATOR TESTS WERE USED TO ESTABLISH SYSTEM CONSTANTS AND TRANSFER FUNCTIONS OF THE COMPONENTS INSTALLED IN THE MISSILE. FIGURE 22 SHOWS THE LOAD TEST ARRANGEMENT USED TO SIMULATE ACTUAL LOADS ON THE FOUR JET ACTUATING CYLINDERS OF THE MISSILE. INERTIA LOADS, W, ARE EQUIVALENT TO THE JET INERTIAS, WITH THE SPRING LOADS ON THE YAW CONTROLS EQUIVALENT TO AERODYNAMIC TRIM TAB LOADS. THE BLOCK DIAGRAM SHOWN IN FIGURE 23 IS A DETAIL BREAKDOWN OF ONE CHANNEL OF THE SYSTEM.

STABILIZATION SYSTEM CONSTANTS ARE GIVEN IN THE FOLLOWING TABLE WITH REFERENCE TO FIGURES 22 AND 23.

MX-774 STABILIZATION SYSTEM CONSTANTS	ROLL	PITCH	YAW
K_1 — VOLTS/DEGREE, FREE GYRO	0.13	0.13	0.13
K_2 — VOLTS/DEGREE/SECOND, RATE GYRO	0.07	0.10	0.10
K_3 — VOLTS/DEGREE JET MOVEMENT, (POSITION FEEDBACK)	0.07	0.04	0.04
K_4 — VOLTS/DEGREE-SECOND, FREE GYRO	0.35	0.30	0.30
K^+ — MILLIAMPS/VOLT, AMPLIFIER	400	400	400
MAXIMUM AMPLIFIER OUTPUT, MILLIAMPS	60	60	60
C — DEGREE/SECOND JET VELOCITY/MILLIAMP	1.4	1.3	1.3
MAXIMUM JET ANGULAR VELOCITY, DEGREES/SECOND	55	50	50
MAXIMUM JET MOVEMENT PER FUNCTION, DEGREES (PITCH AND ROLL OR YAW AND ROLL MAY ADD)	5	5	5
TORQUE ON MISSILE/DEGREE JET, TAKE OFF COND, FT LB	80	815	815
MOMENT OF INERTIA OF MISSILE, MAXIMUM, SLUG-FT^2	100	3900	3900

ADDITIONAL TESTS HAVE BEEN RUN ON ROLL SYSTEM STABILIZATION, IN WHICH THE RATIO BETWEEN FREE GYRO AND RATE GYRO OUTPUTS WAS CHANGED. FOR THE FREE GYRO "K_1", AS SHOWN IN THE ABOVE TABLE, WAS CHANGED FROM 0.13 TO 0.07 VOLTS PER DEGREE, AND FOR THE RATE GYRO "K_2" WAS CHANGED FROM 0.07 TO 0.04 VOLTS PER DEGREE PER SECOND RATE. THESE CHANGES RESULTED IN THE SYSTEM BEING CRITICALLY DAMPED IN ROLL UNDER TAKE-OFF CONDITIONS. FIGURE 24 SHOWS RESPONSE OF THE ROLL SYSTEM ON THE SIMULATOR AFTER INTRODUCTION OF A STEP FUNCTION ERROR SIGNAL.

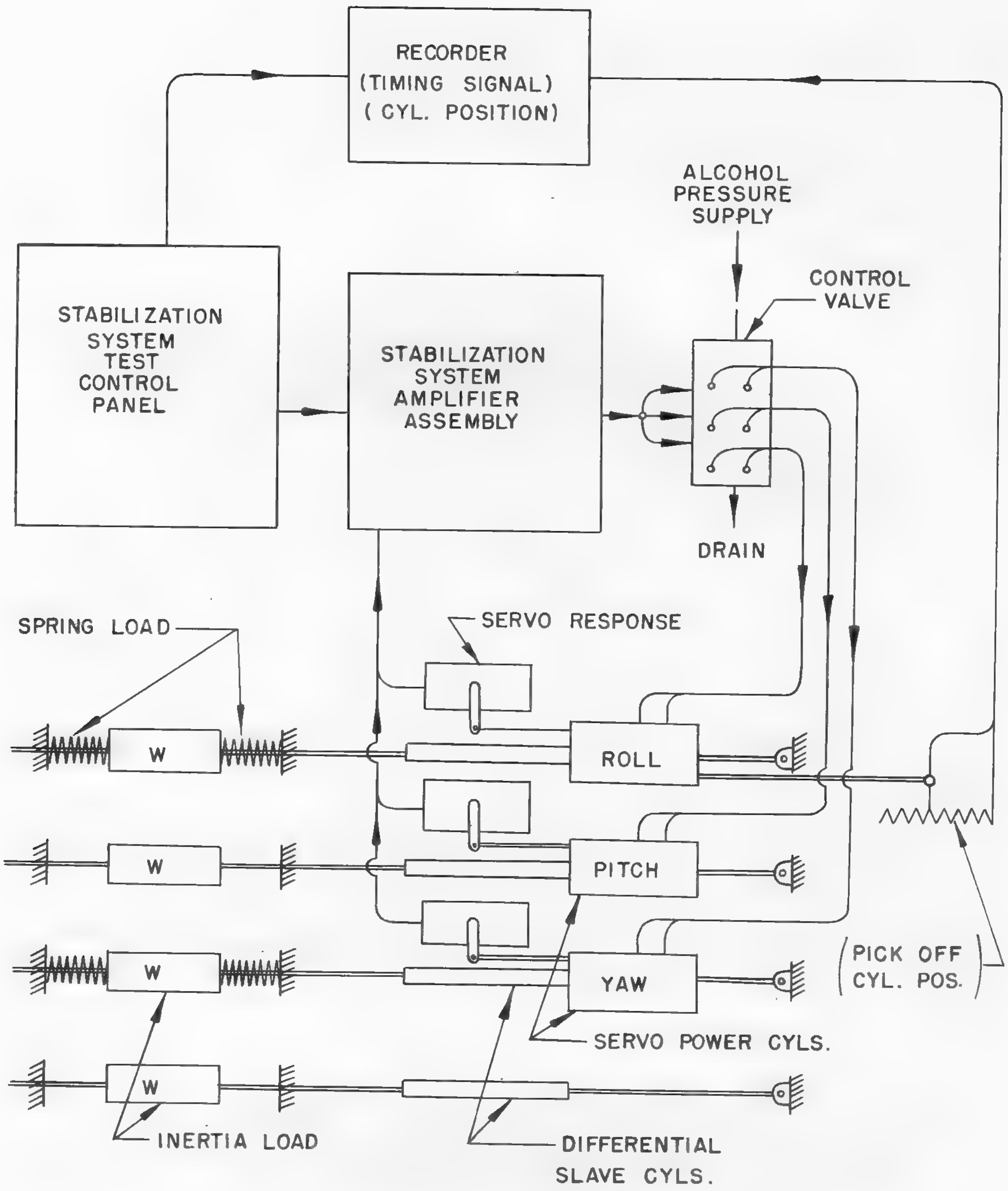

FIG 22 -- MX-774 SERVO SYSTEM LOAD TEST

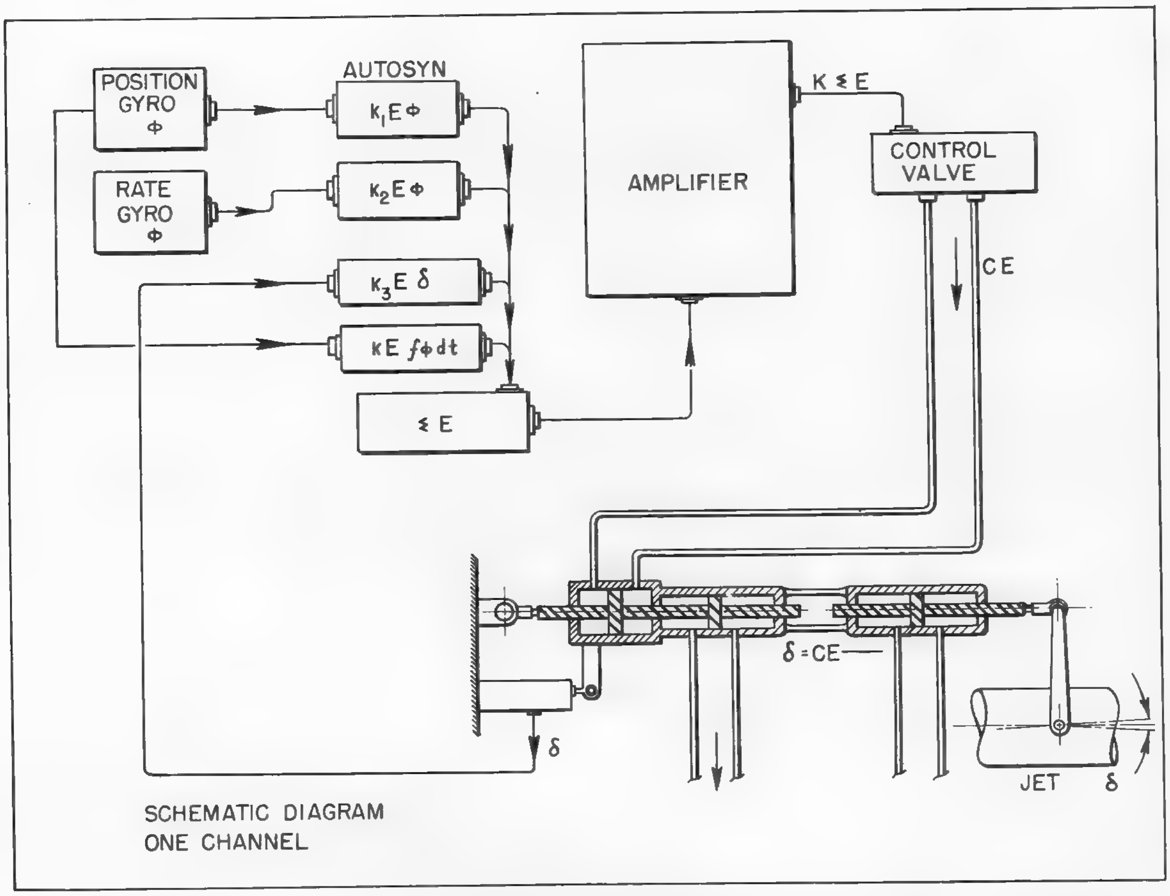

FIG 23 -- SERVO SYSTEM LOAD TEST SCHEMATIC OF ONE CHANNEL

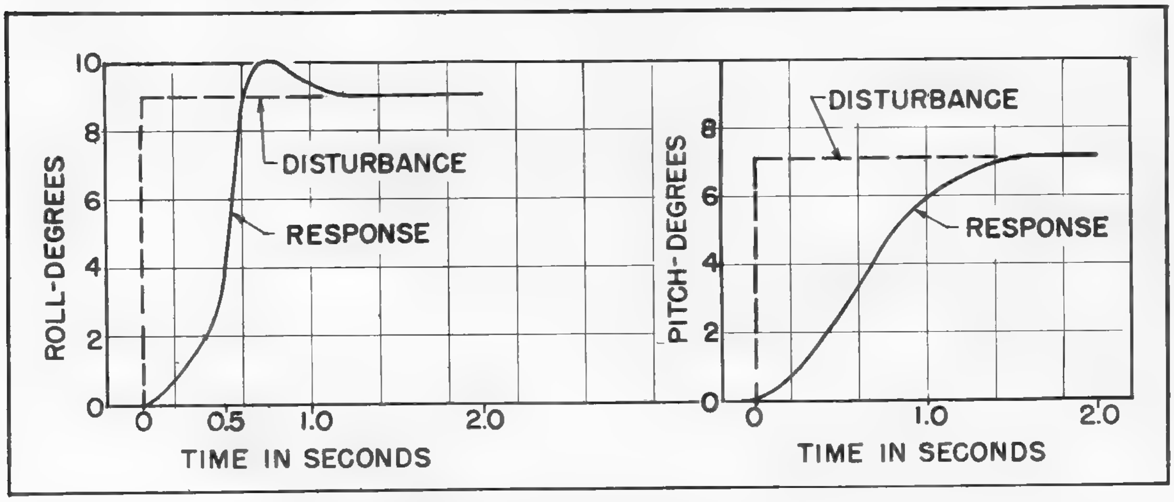

FIG 24 -- STEP RESPONSE OF STABILIZATION SYSTEM SIMULATOR

Dynamic tests were run with the stabilization system controlled through the special test panel shown in Figure 25. Figure 26 shows the results in phase lag and decrease in amplitude of servo cylinder travel with increase in frequency. This test was run with the system on the load test arrangement diagrammed in Figure 22. Tests were also run with the test panel connected to the system within the missile during both hot and cold ground runs.

C. TEST EQUIPMENT

All special test equipment required for ground tests and for missile firing has been completed, ready for use, and includes the following items:

(1) Stabilization system test control panel
(2) Alcohol pressure supply unit
(3) Stabilization amplifier test unit
(4) Integral control amplifier test unit
(5) Solenoid valve power unit (for test and adjustment)

Figure 27 shows the portable alcohol pressure supply unit designed for use in ground tests on the stabilization system when the missile fuel pumps are not operating. The accumulator is pressurized with nitrogen to 400 pounds per square inch after being half-filled with alcohol. An automatic pressure switch connected to the pump motor maintains pressure during test runs.

Step function and dynamic tests were run on the stabilization system with the special test panel arrangement. Results were recorded, so that system performance could be analyzed with Figure 24 as a typical example of the results. Duplicate tests were run, recording first with the Brush Recorder and then by telemetering and a Consolidated Vultee recorder, in order to eliminate minor operational defects in the system.

Practice runs were made with the flight program drive unit in control. In actual ground tests, however, this unit was disconnected and the test program timer in the test panel was used. Results of practice runs and the firing test show satisfactory operation of the automatic test control unit.

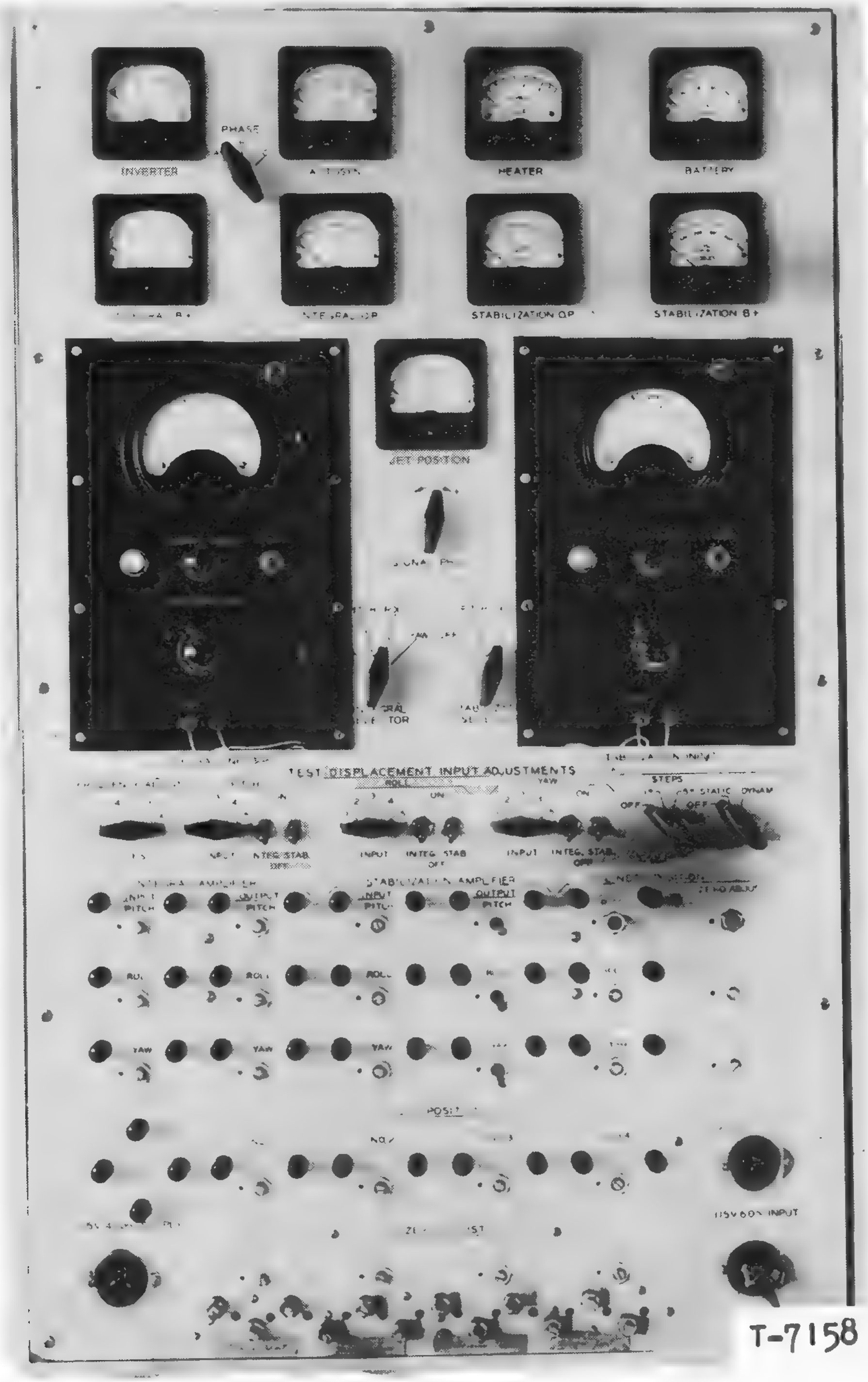

FIG 25 -- MX-774 STABILIZATION SYSTEM TEST PANEL

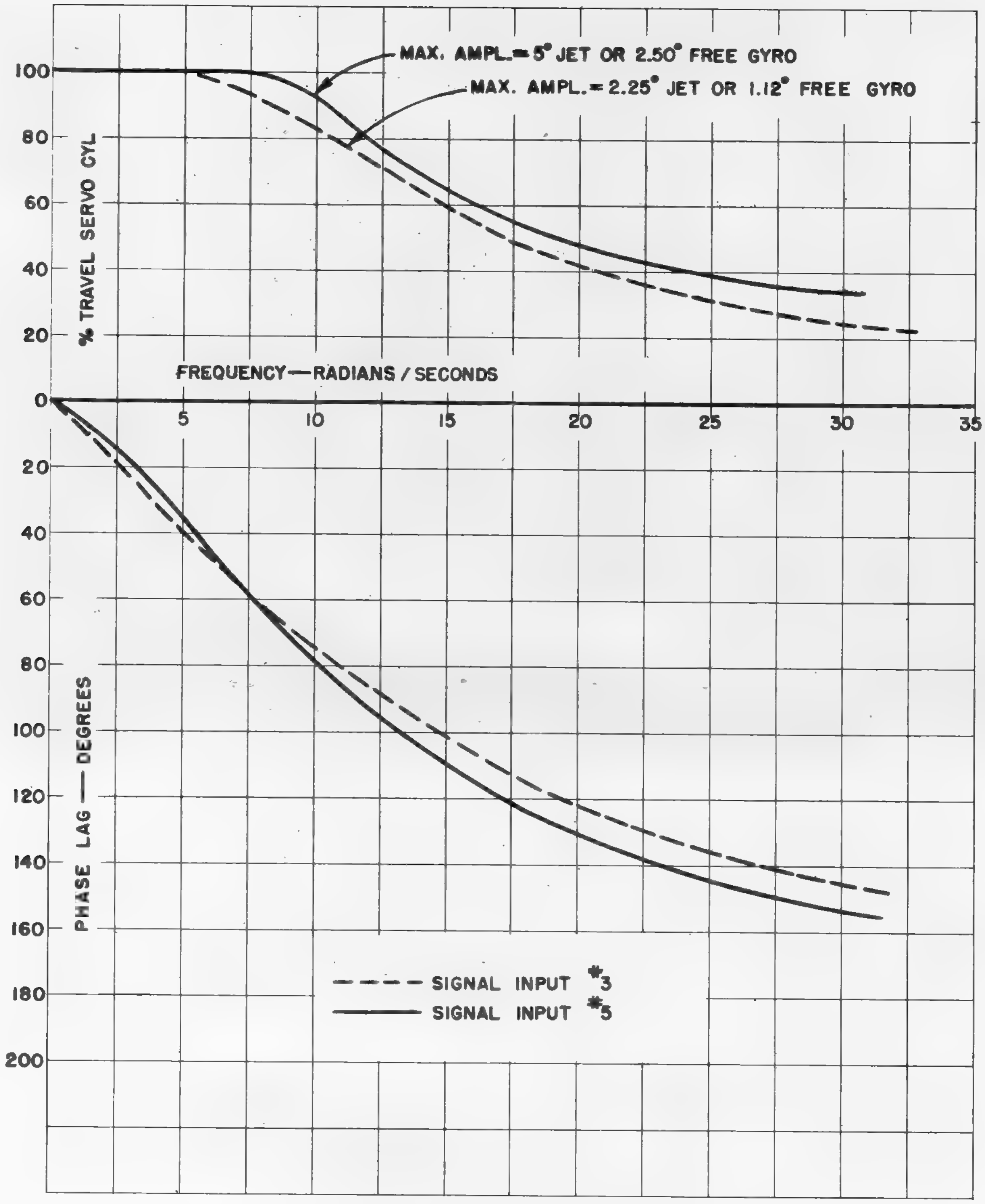

FIG 26 -- PHASE LAG AND SERVO CYLINDER TRAVEL WITH FREQUENCY

FIG 27 -- ALCOHOL PRESSURE SUPPLY CART

SECTION VII
GUIDANCE AND INTELLIGENCE

A. VHF POSITION TRACKER

A FOURTH GROUND STATION RECEIVER HAS BEEN CONSTRUCTED AND TESTED WHICH, WITH THE THREE RECEIVERS PREVIOUSLY CONSTRUCTED, PROVIDES FOUR GROUND STATION POSITION TRACKING RECEPTION CHANNELS FOR THE FIRST MISSILE APPLICATION. TWO OF THESE RECEIVERS ARE TO BE USED IN AZIMUTHAL POSITIONING OF THE MISSILE WITH THE REMAINING TWO TO BE STANDBYS. ACCURACIES AS HIGH AS ONE TO TWO DEGREES OVER THE ENTIRE INPUT RANGE OF 70 DB AND A FREQUENCY RANGE OF 400 KC ARE NOW BEING ACHIEVED ON ALL FOUR RECEIVER CHANNELS. VIEWS OF A COMPLETED RECEIVER ARE SHOWN IN FIGURE 28.

THE PHASE COMPARER, DESIGNED TO MEASURE THE PHASE DIFFERENCE BETWEEN SIGNALS OF THE TWO AZIMUTH CHANNELS AND TO FURNISH THE INFORMATION IN SUITABLE FORM FOR THE COMPUTER, HAS UNDERGONE FINAL CALIBRATION AND OPERATIONAL TESTS. CONSIDERABLE ALIGNMENT AND ADJUSTMENT OF GEAR TRAINS WAS NECESSARY TO PROVIDE COMPLETELY SATISFACTORY PERFORMANCE. THE FINISHED UNIT IS SHOWN IN FIGURE 29. A PHASE-CALIBRATOR HAS BEEN CONSTRUCTED TO AID IN CALIBRATION OF THE PHASE COMPARER.

WORK HAS BEEN COMPLETED ON DEVELOPMENT AND CONSTRUCTION OF A MANUALLY OPERATED COMPUTER FOR USE IN FLIGHT TESTS OF THE SYSTEM WITH THE C-46 AIRPLANE. CONSTRUCTION OF AN AUTOMATIC COMPUTER FOR USE WITH THE MISSILE HAS ALSO BEEN COMPLETED. IT IS PLANNED THAT ANY DEVIATION FROM THE PRECOMPUTED COURSE BE DETECTED BY THE POSITION TRACKING CIRCUITS AND AN AUTOMATIC CORRECTION BE APPLIED BY THE COMPUTER THROUGH THE COMMAND CONTROL SYSTEM.

ADDITIONAL OPERATIONAL TESTS HAVE BEEN CONDUCTED ON THE No. 2 AND No. 3 PRESSURIZED VHF TRANSMITTERS. THESE UNITS EMPLOY AN INTEGRAL DYNAMOTOR AS POWER SUPPLY AND PROVIDE A POWER OUTPUT OF 50 WATTS C-W AT A FREQUENCY OF 150 MEGACYCLES. VIEWS OF ONE OF THE TRANSMITTERS IN A TEMPORARY MOUNT OUTSIDE ITS PRESSURIZED CONTAINER (AS USED IN FIELD TESTING) ARE SHOWN IN FIGURE 30.

T-7242

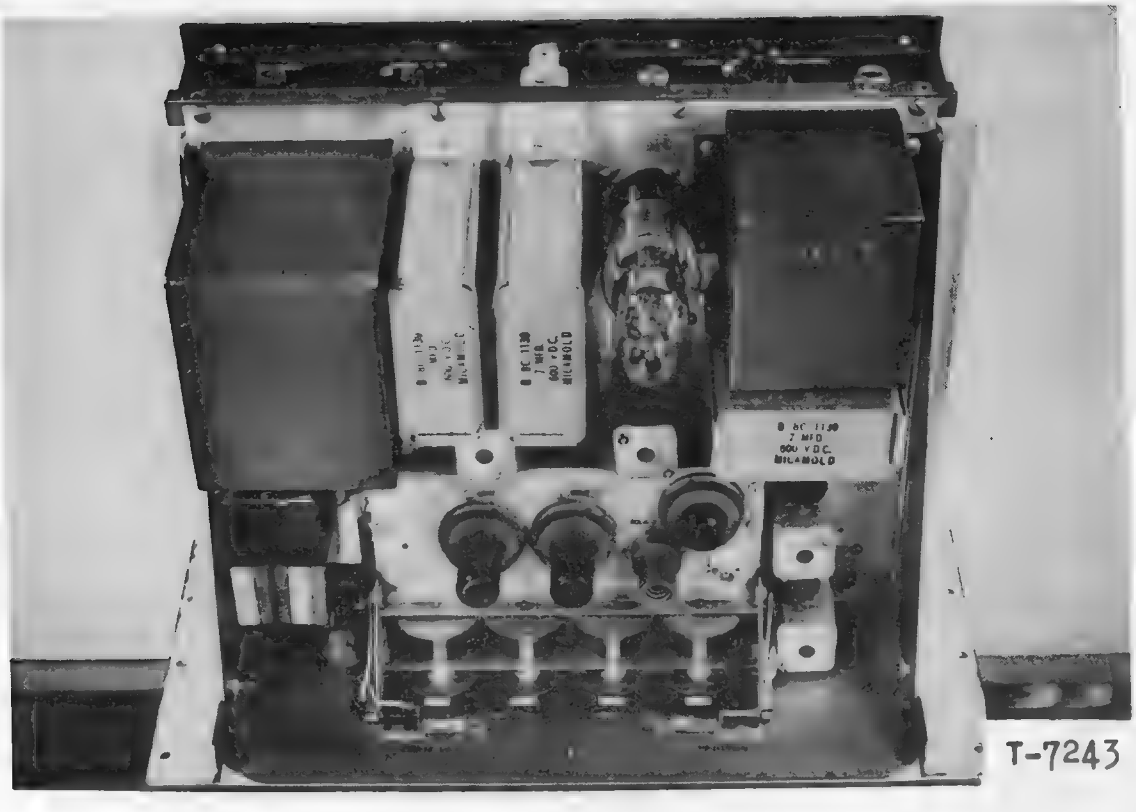

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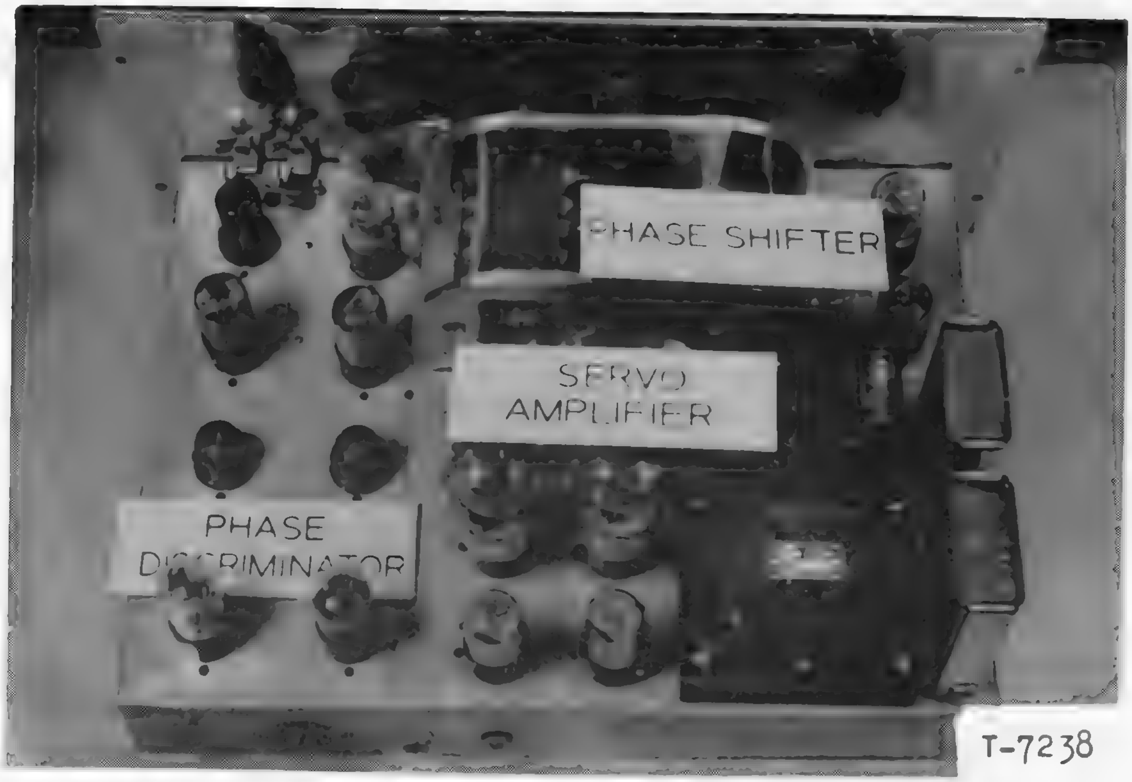
FIG 28 -- POSITION TRACKER RECEIVER



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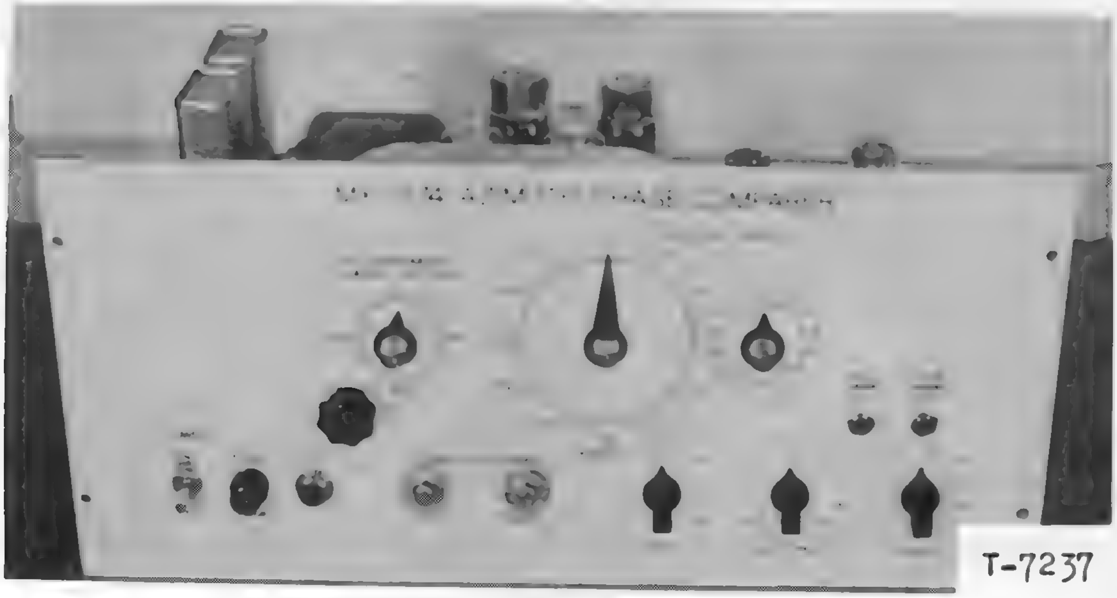

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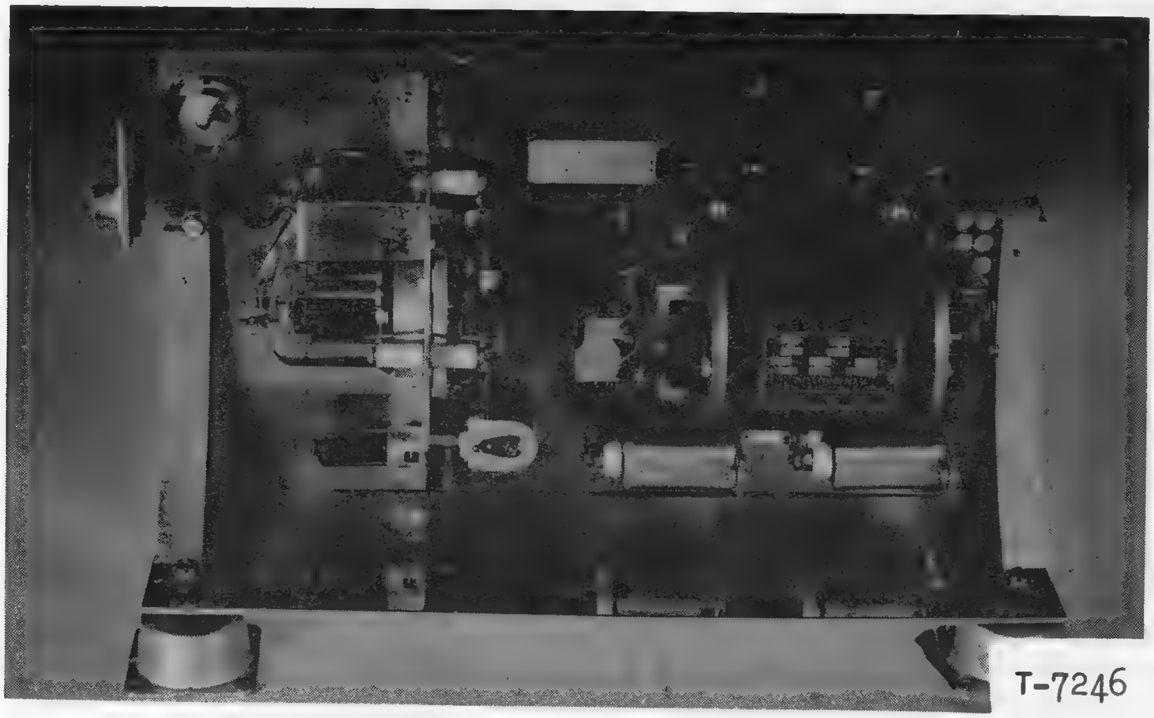
FIG 29 -- AZIMUTH PHASE COMPARER



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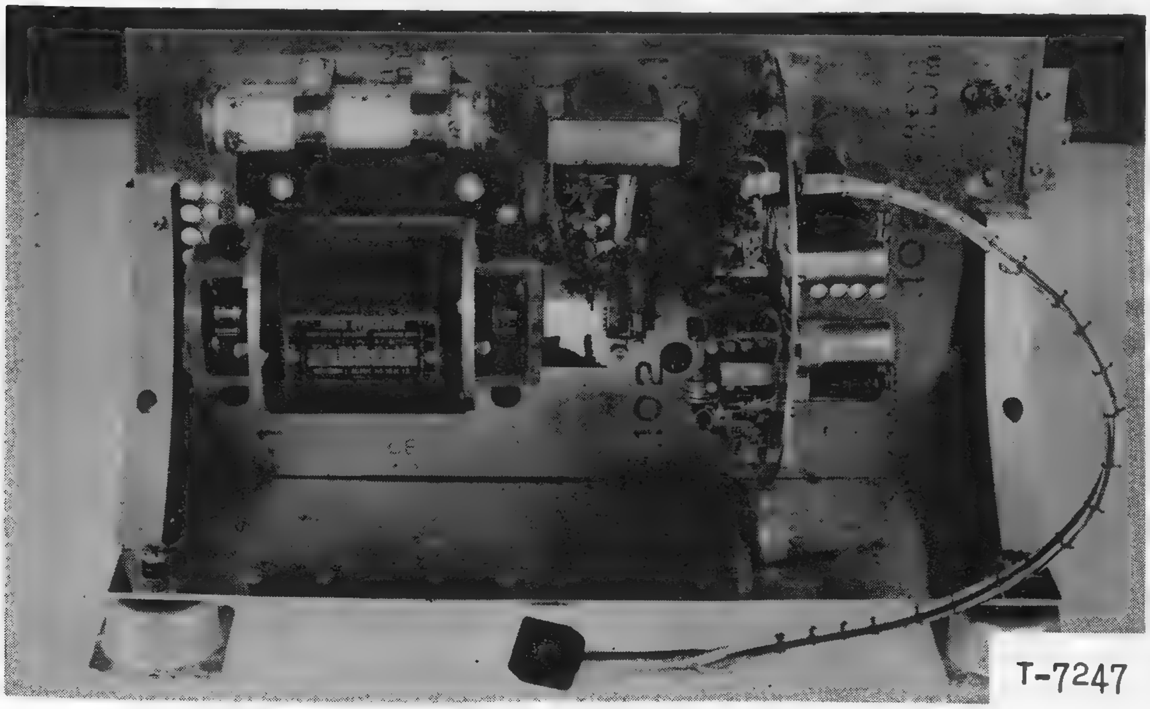

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FIG 30 -- FINAL AMPLIFIER AND OSCILLATOR DECKS OF POSITION TRACKER TRANSMITTER

THE POSITION TRACKER GROUND STATION EQUIPMENT, CONSISTING OF AN AZIMUTH PHASE COMPARER, FOUR RECEIVERS (TWO BEING SPARES), A LOCAL OSCILLATOR UNIT, AND AN AUXILIARY POWER SUPPLY, HAS BEEN MOUNTED IN A CABINET HAVING AN EXTERNAL MONITORING JUNCTION BOX AS SHOWN IN FIGURE 31.

THE FIRST FIELD TEST OF THE CONSOLIDATED VULTEE PHASE COMPARISON TRACKING METHOD HAS BEEN CONDUCTED WITH GRATIFYING RESULTS. NO SERIOUS OPERATIONAL DIFFICULTIES WERE ENCOUNTERED WITH THE SYSTEM. IN ORDER TO CONDUCT THE TEST, TWO YAGI-TYPE ANTENNA ARRAYS WERE SET UP 200 FEET APART WITH THE POSITION TRACKER EQUIPMENT LOCATED EXACTLY HALFWAY BETWEEN. ONE OF THE TRANSMITTERS CONSTRUCTED FOR USE IN THE MX-774 MISSILE WAS SET UP WITH A DUMMY LEAD ON A HAND-DRAWN CART PLACED AT A LOCATION 300 FEET FROM A POINT MIDWAY BETWEEN AND EQUIDISTANT FROM THE TWO ANTENNAS, AS SHOWN IN FIGURE 32. MOTION OF THE CART WAS INDICATED ON THE DIAL OF THE AZIMUTH PHASE COMPARER. EQUIPMENT READINGS OBTAINED DURING THIS FIRST ROUGH TEST WERE REPEATED WITHIN THE ACCURACY OF PLACEMENT OF THE TRANSMITTER CART. REFLECTIONS FROM CARS AND PERSONS SEEMED TO AFFECT THE PHASE READINGS WHEN THEY WERE WITHIN ABOUT 100 FEET OF EITHER OF THE ANTENNAS OR BETWEEN THE TRANSMITTER CART AND ONE OF THE RECEIVING ANTENNAS.

A SIMILAR SET-UP WAS MADE WITH THE TRANSMITTER AT A POINT 240 FEET FROM THE ANTENNA BASELINE. RESULTS OF THIS TEST DISCLOSED THAT A ONE-INCH MOVEMENT OF THE TRANSMITTER CART AT A DISTANCE OF 240 FEET FROM THE ANTENNA BASELINE PROVIDED ABOUT 5 DEGREES MOTION IN THE DIAL OF THE PHASE COMPARER, AND THAT 1/4-INCH MOVEMENT OF THE CART AT 240 FEET WAS EASILY DETECTABLE.

A FLIGHT TEST OF THE POSITION TRACKER EQUIPMENT WAS PERFORMED DURING THE SUBJECT REPORT PERIOD. A SMALL PLANE, CONTAINING A 150 MEGACYCLE TRANSMITTER, WAS FLOWN ON A SEAWARD ROUTE FROM A TRACKING SITE ON POINT LOMA. FLYING ON A GIVEN COMPASS HEADING, IT WAS TRACKED OPTICALLY WITH A THEODOLITE AND WITH THE POSITION TRACKING EQUIPMENT. THE RESULTS SHOWED THE DEVIATION OF THE ELECTRICALLY MEASURED COURSE FROM THE SAME COURSE AS MEASURED BY OPTICS TO BE LESS THAN 0.3 DEGREES WHICH WAS WITHIN THE OPTICAL SYSTEM TRACKING ERRORS. MORE EXTENSIVE FLIGHT TESTS ARE PLANNED FOR THE SYSTEM USING A C-46 AIRPLANE TO SIMULATE THE MISSILE.

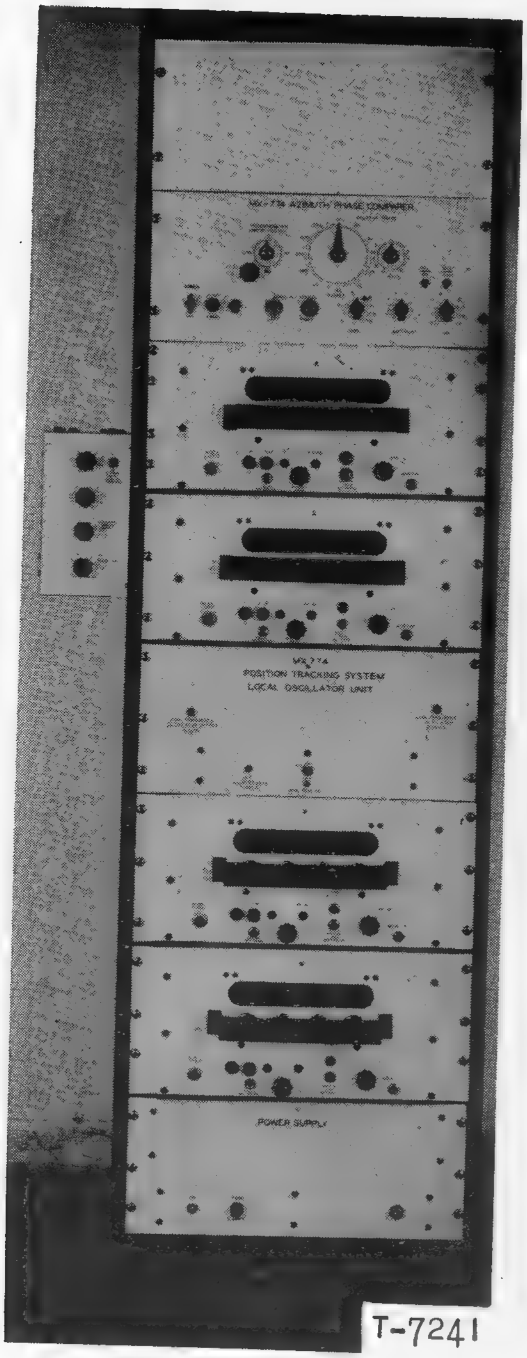


FIG 31 -- POSITION TRACKER EQUIPMENT

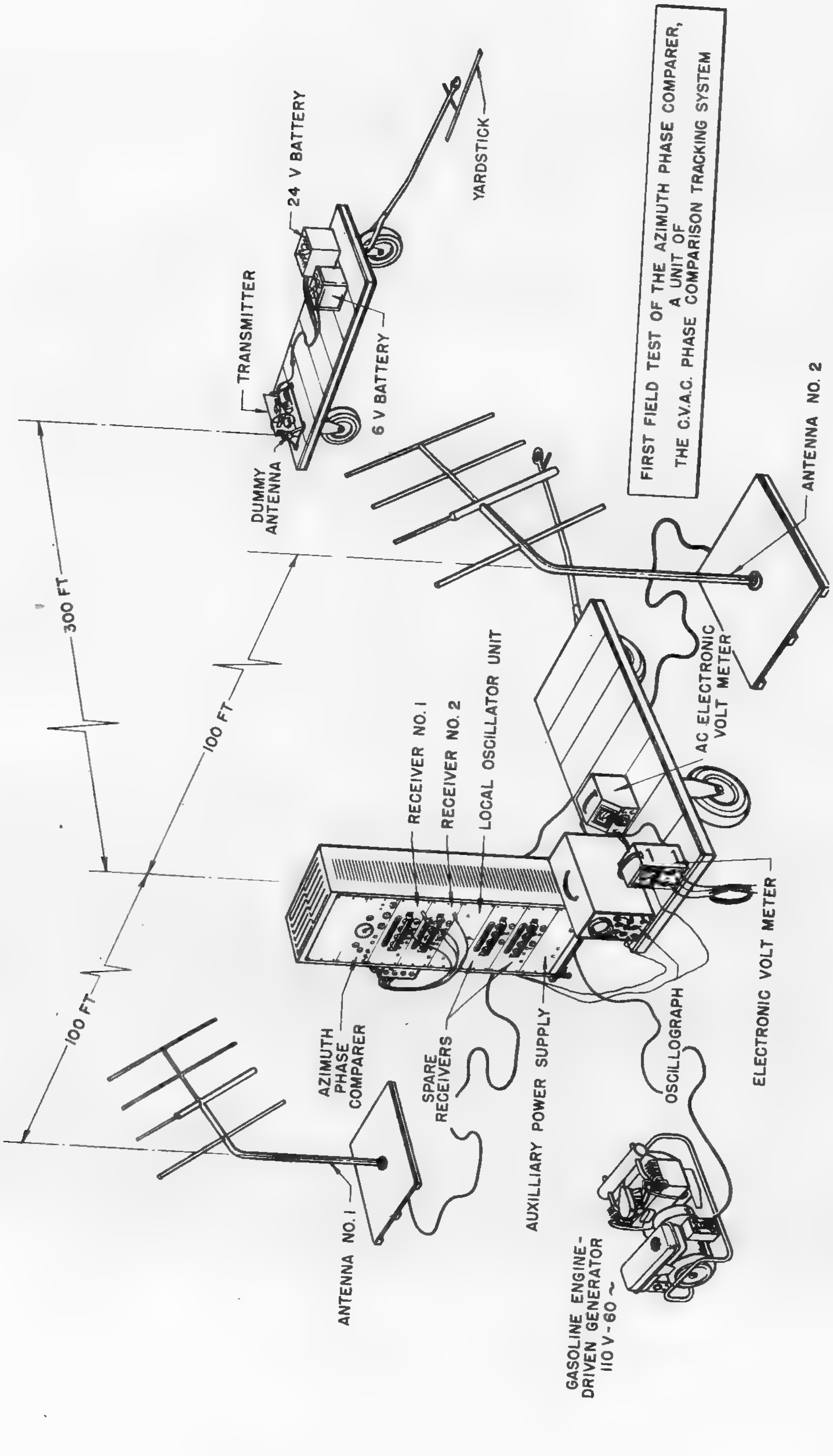


FIG 32 -- FIELD TEST SET-UP OF AZIMUTH PHASE COMPARER

B. ANTENNA DESIGN AND CONSTRUCTION

Tests have been conducted to determine the type, and location on the missile, of a suitable airborne transmitting antenna to be used with the phase comparison position tracking system. The results of these tests have been issued in Reference 6 and are summarized below.

The first effort at design of a flush mounting type of missile antenna resulted in construction of a flush type eighth-wave radiator utilizing a slot opening. Fair radiating efficiency was achieved but the input impedance was found to be extremely high and radiation was almost entirely in a forward direction. Since this was directly opposed to requirements stipulated for the antenna, the design was abandoned.

An external type of antenna, Figure 33, mounted within an exposed portion of dummy fairing situated upon and parallel to a vertical fin on the Test Vehicle was found more satisfactory for the purpose intended. Radiation was predominately in a rearward direction from the Test Vehicle with no sharp deviations of the pattern being observed in the field strength radiated in that direction. The input impedance of the antenna was found to be 300-ohms and was satisfactorily coupled to a 52-ohm coaxial cable by means of a small lumped-circuit matching network.

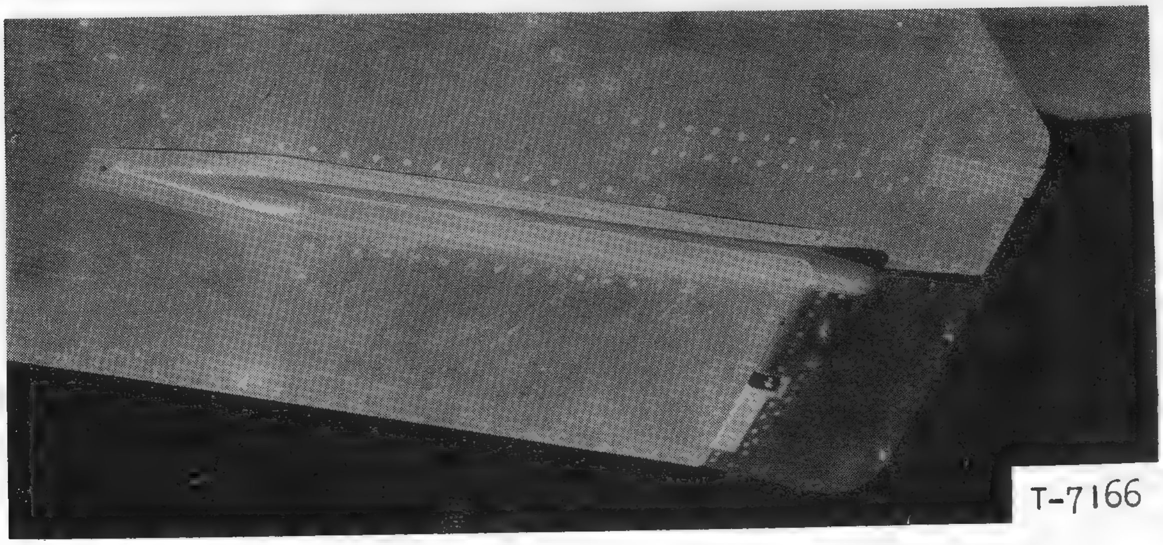


FIG 33 -- MISSILE ANTENNA INSTALLATION FOR POSITION TRACKING

Aerodynamically, the dummy fairing and its associated Textolite rod which supports the wire antenna have been found to produce a minimum of drag. Specifications of the Textolite rod employed with the antenna are such as to insure its enduring the range of temperatures expected to be encountered.

C. DOPPLER SPEEDOMETER EXPERIMENTS AND DESIGN

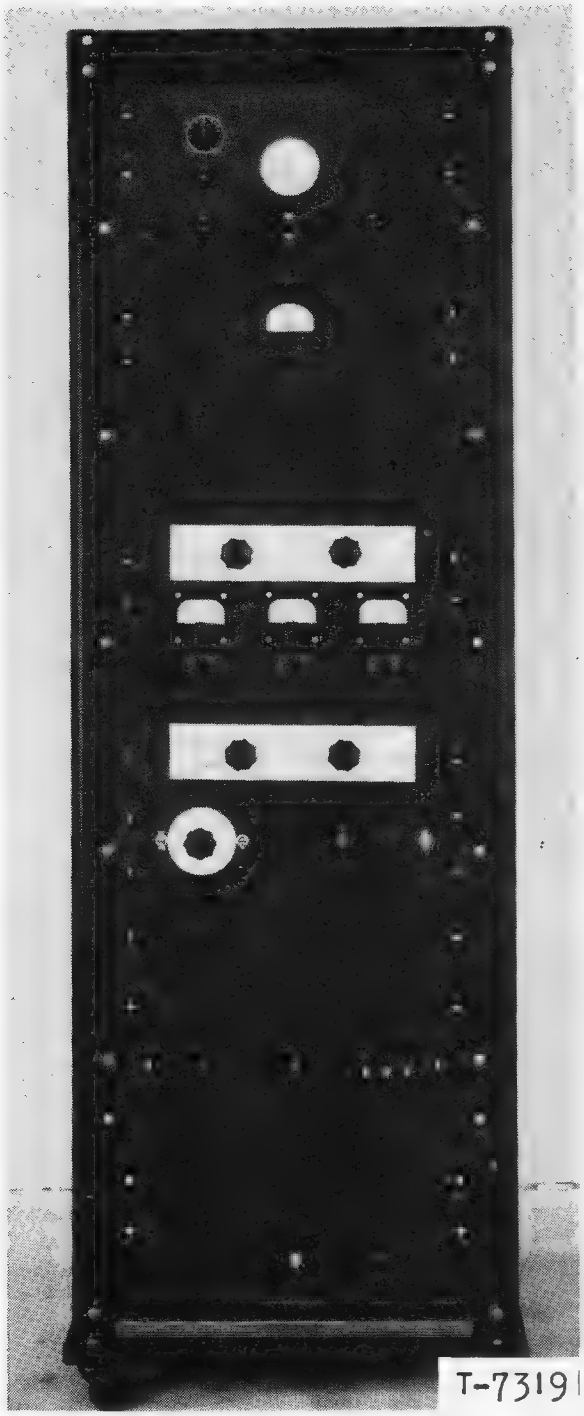
Work on this program consisted of redesign of the oscillator section of the ground station transmitter and completion of construction and tests of the second production type superheterodyne repeater station.

Construction of the frequency demodulator unit was discontinued during the period in favor of a change in the oscillator section of the ground station transmitter. A new oscillator unit for the ground station transmitter has been designed in which two crystal controlled frequencies are generated; one is phase modulated (to provide range intelligence) whereas the other is not modualted and provides the Doppler velocity intelligence. Use of this system requires no modification of any of the present existing Doppler equipment and appears to offer the most satisfactory solution to the problem of reducing the effects of frequency modulation on the Doppler speedometer readings.

Completed during the subject report period, the second production type superheterodyne repeater station has been given final tests to determine its operational characteristics. The tests disclosed that an input of 10 microvolts to the repeater station produces an output of 10 watts at the second harmonic of the 72-megacycle input signal.

D. RANGE MEASUREMENT SYSTEM

All of the measurement units of the range measurement system have been completed and are in the process of clean-up and testing. This ground station equipment, with exception of the units which are an integral part of the Doppler speedometer system, has been assembled in a cabinet as shown in Figure 34. To facilitate operation and testing of the range measurement system, a monitoring cathode ray oscilloscope has been built into the relay rack as an integral part of the equipment.



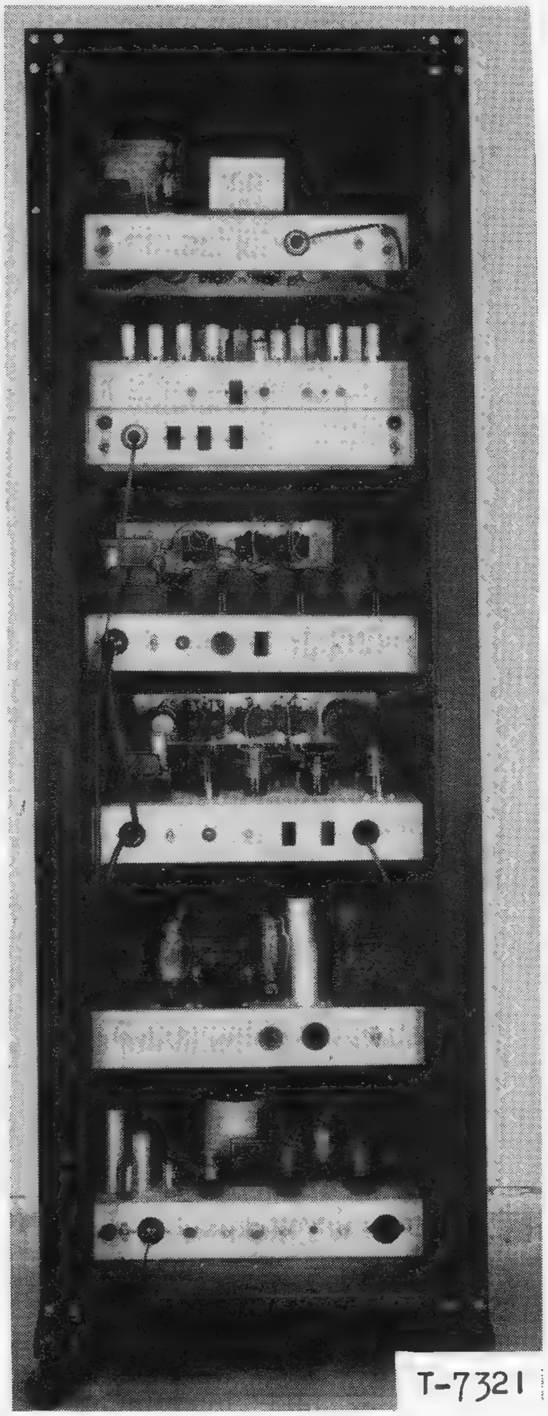
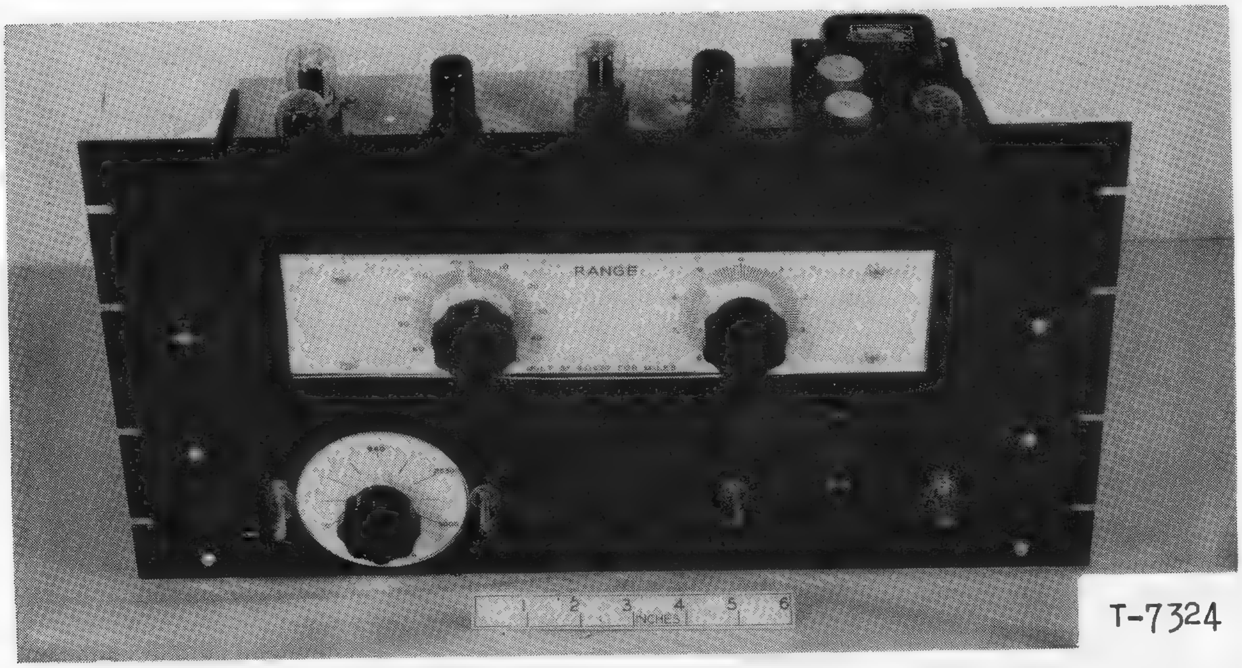


FIG 34 -- RANGE MEASUREMENT EQUIPMENT

A phase rotating simulator unit, Figure 35, has been constructed for testing the range measuring equipment before actual use in missile range measurement. This simulator incorporates a gear train driven, by a variable speed motor, from zero to equivalent speeds well above the highest speed of the missile. Connected to each of 3 separate gears in the gear train is a phase rotating transformer. The gears are so proportioned in the train as to allow 3 operating frequencies to rotate in phase at a ratio of 1 to 25 to 125. The operating frequencies, originated in a crystal controlled generator unit, Figure 36, are of 800 cycles, 20,000 cycles and 10,000 cycles. This last is derived from the 20,000 cycle signal by heterodyne and frequency multiplication action.

A switch is provided on the simulator to allow either of two methods of operation. The first, in which only the 800 cycle and the 20,000 cycle signals are rotated in phase by the simulator, is an exact simulation of the operation of the range measurement system when actually employed in measuring distance. However, due to the lack of absolute linearity of the phase rotating transformers, this method of operation results in an irregular follow-up action of the range indicator. The second method of operation involves use of the 10,000 cycle phase rotating transformer. The 10,000 cycle signal, previously derived from the 20,000 cycle signal after passing through its phase rotating transformer, is by-passed. In its place a 10,000 cycle signal from the signal generator is directed through the phase rotating transformer in the simulator and applied to a servo-controlled phase discriminator in the range indicator unit. This action gives an improvement in linearity of follow-up and permits greater precision in testing.

Another switch in the simulator circuit interrupts the gear-driven signals to the range measurement equipment and is equivalent to momentarily losing the signal from the missile. This switch therefore allows testing of the over-ride system to ensure its capacity to "catch-up" when the signal is restored after an interruption. Because of friction in the computing gear train, inertia in the rotor driving the gear train, and electrical inertia in the system due to time constants, the computing gear train can never rotate in perfect synchronism with the applied signals but may be made to run more smoothly by application of a rate-of-change error signal to the appropriate portion of the electrical or mechanical circuit.



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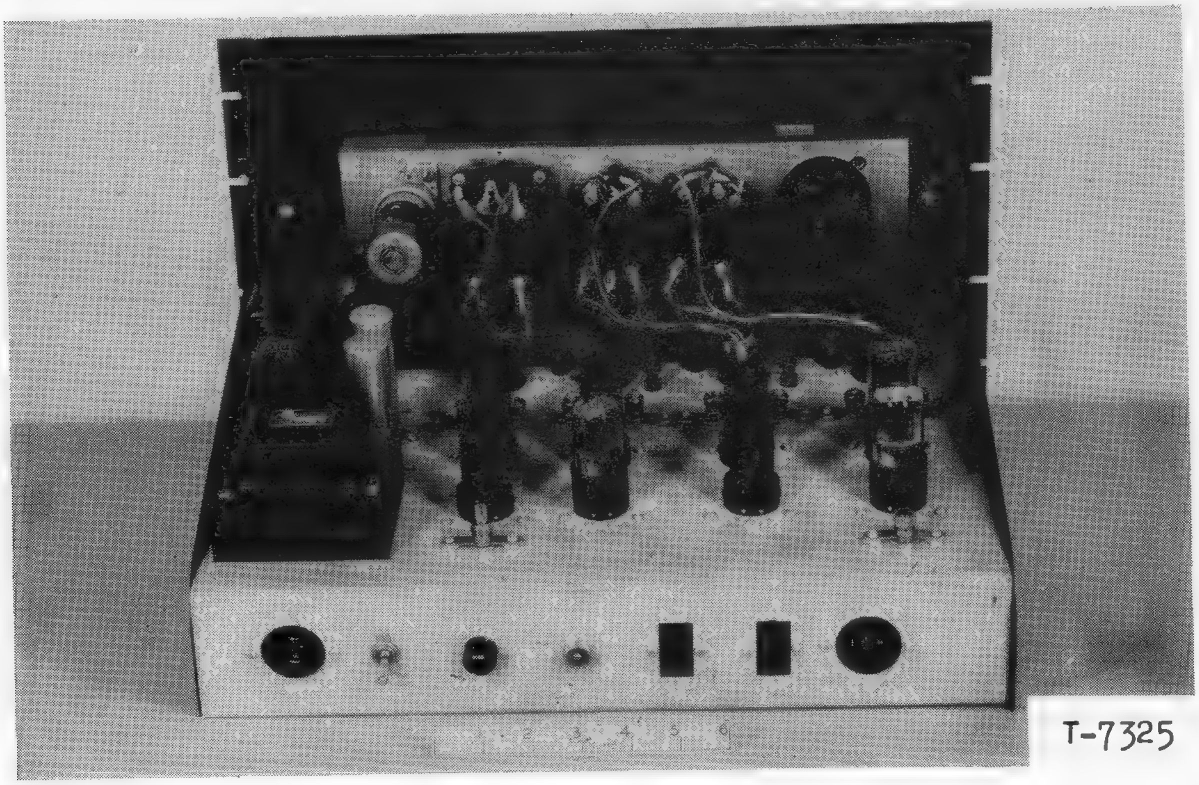

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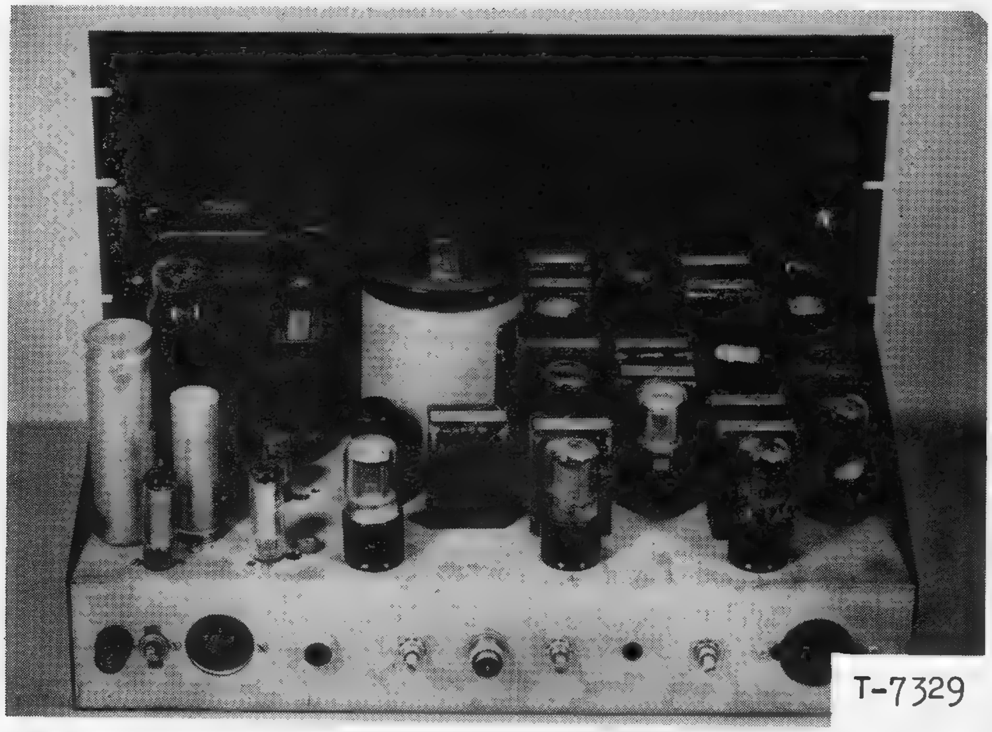
FIG 35 -- RANGE SIMULATOR FOR PRE-OPERATION TESTS



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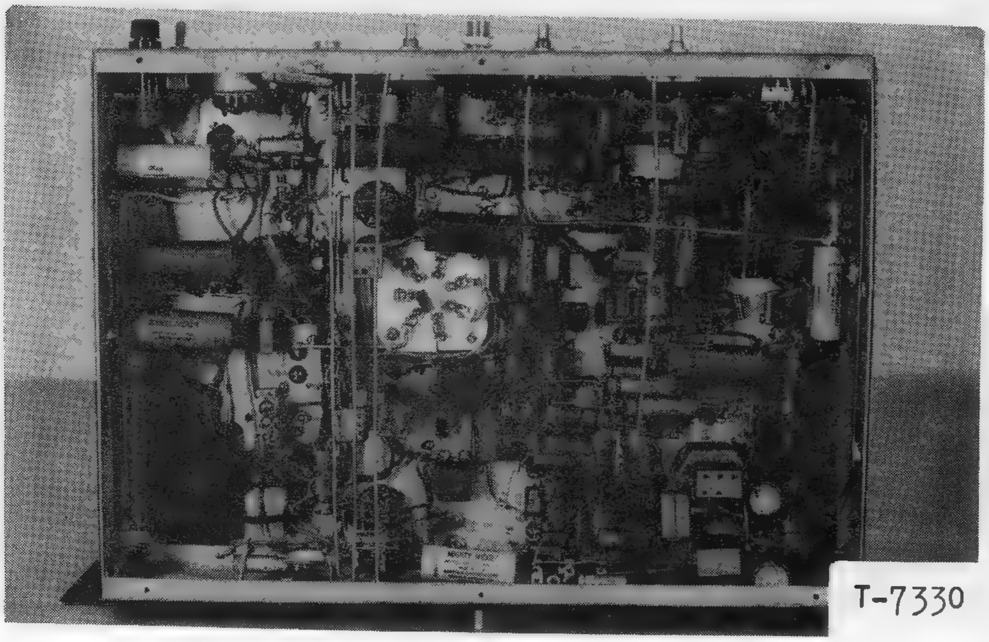

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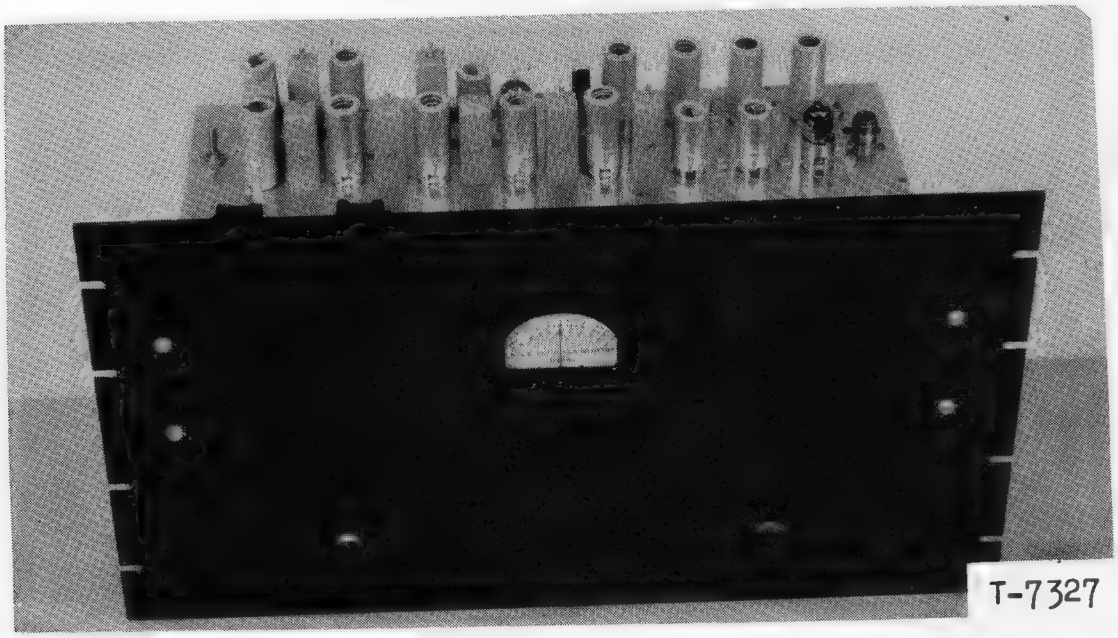
FIG 36 -- RANGE MEASUREMENT FREQUENCY GENERATOR



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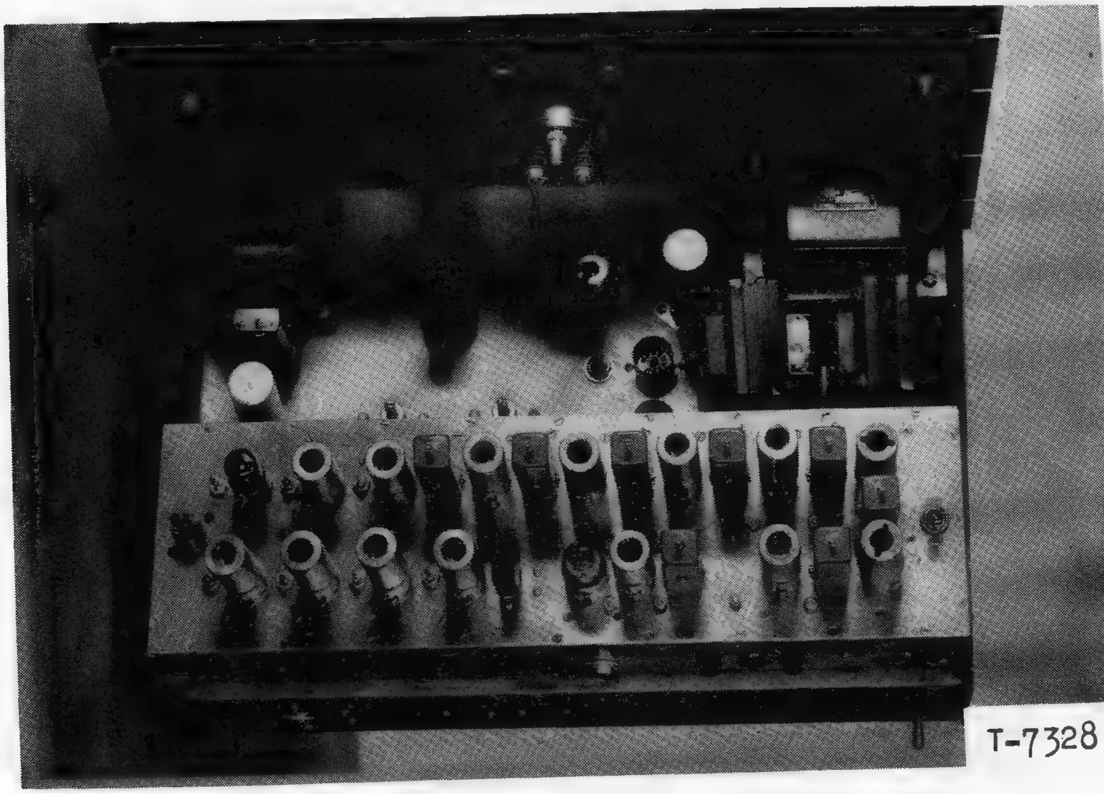

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FIG 37 -- FREQUENCY MODULATION TYPE RANGE SYSTEM RECEIVER

A DERIVATIVE CIRCUIT WAS ADDED TO THE SERVO LOOP TO STABILIZE OVERALL OPERATION OF THE SYSTEM. LAG AND HUNT BETWEEN THE RANGE INDICATOR AND THE "POSITION" OF THE SIMULATOR HAVE BEEN REDUCED FROM 300 FEET TO 25 FEET.

THE FREQUENCY MODULATION TYPE RECEIVER, EMPLOYED AT THE GROUND STATION TO RECEIVE AND DEMODULATE THE RETURNED SIGNAL RECEIVED FROM THE AIRBORNE REPEATER STATION, HAS BEEN COMPLETED. VIEWS OF THIS UNIT ARE SHOWN IN FIGURE 37.

THE TUNED RADIO FREQUENCY REPEATER FOR USE WITH THE SYSTEM HAS BEEN COMPLETED AND IS IN FINAL STAGES OF ADJUSTMENT AND ALIGNMENT. WITH A SENSITIVITY OF 10 MICROVOLTS, ITS FUNCTIONING HAS PROVEN VERY SATISFACTORY DURING OPERATIONAL TESTS IN THE LABORATORY. THIS UNIT IS SHOWN IN FIGURE 38.

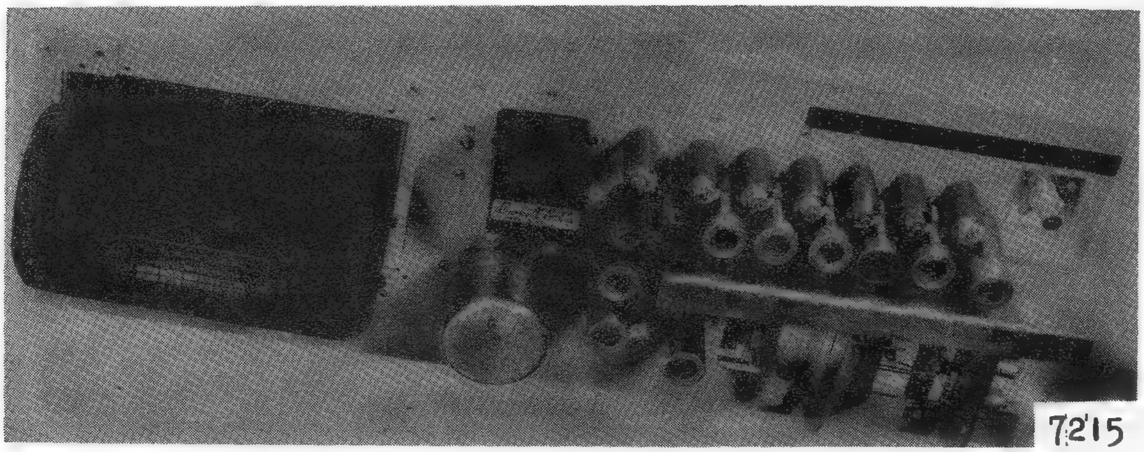
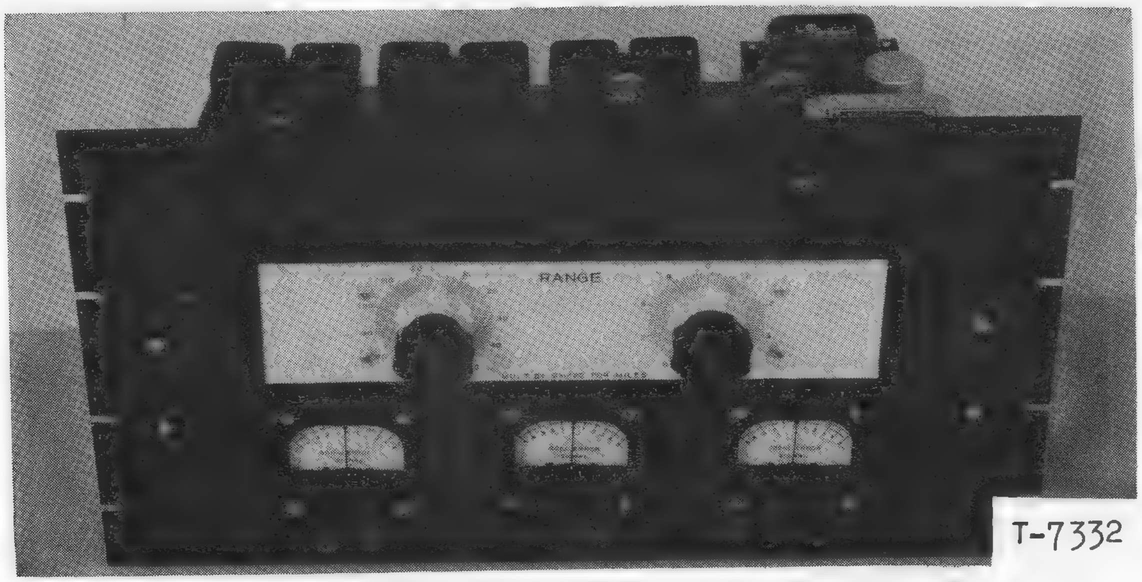


FIG 38 -- DOPPLER AIRBORNE REPEATER

THE REMAINDER OF THE BASIC EQUIPMENT, INCLUDED IN THE GROUND STATION AND COMPLETED DURING THE PREVIOUS WORK PERIOD, HAS UNDERGONE OPERATIONAL TESTS AND FINAL CLEAN-UP. THESE UNITS, THE PHASE SHIFT DISCRIMINATOR UNIT AND THE MOTOR CONTROL UNIT, ARE SHOWN IN THEIR FINAL FORM IN FIGURE 39 AND 40, RESPECTIVELY.

A NEW MODULATOR IS UNDER CONSTRUCTION FOR FREQUENCY MODULATING THE GROUND STATION TRANSMITTER WITH THE COMPLEX WAVE (800 CPS AND 20,000 CPS) WHICH IS USED TO SUPPLY THE TWO REFERENCE FREQUENCIES FOR PHASE MEASUREMENT OF RANGE. CONSTRUCTION OF THIS MODULATOR IS APPROXIMATELY 80 PERCENT COMPLETE.



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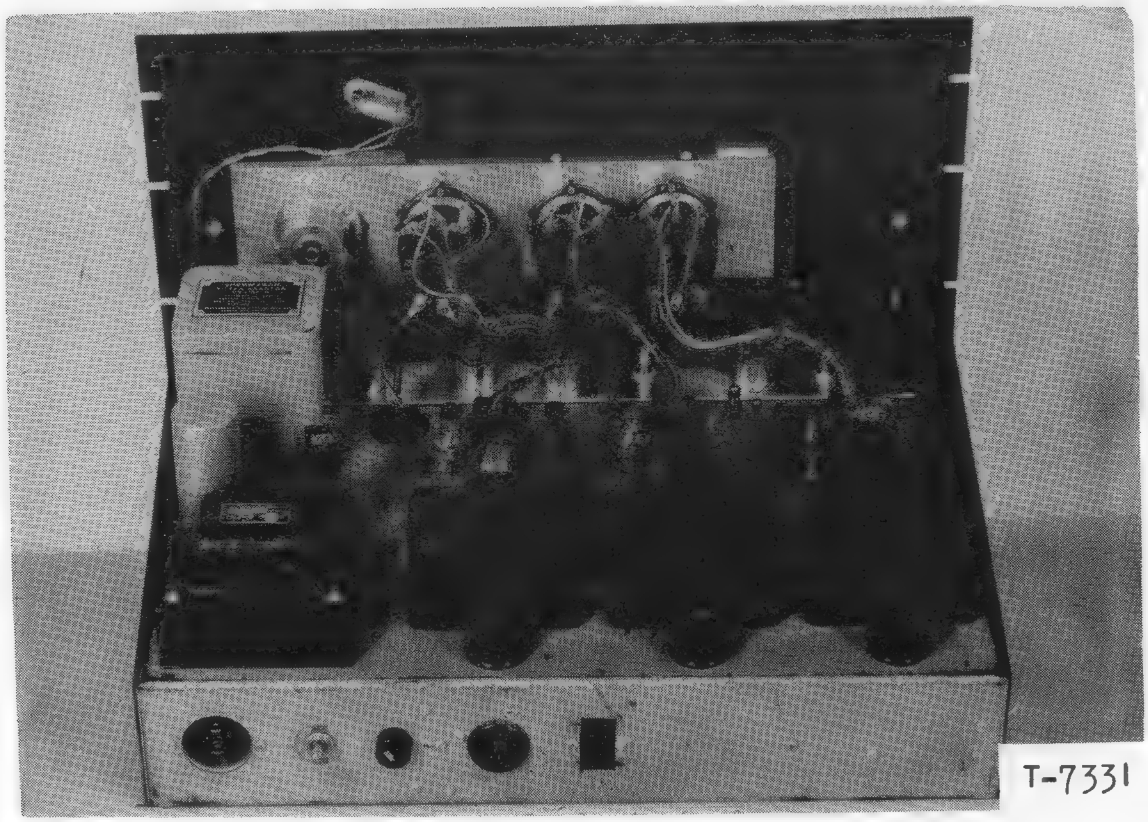

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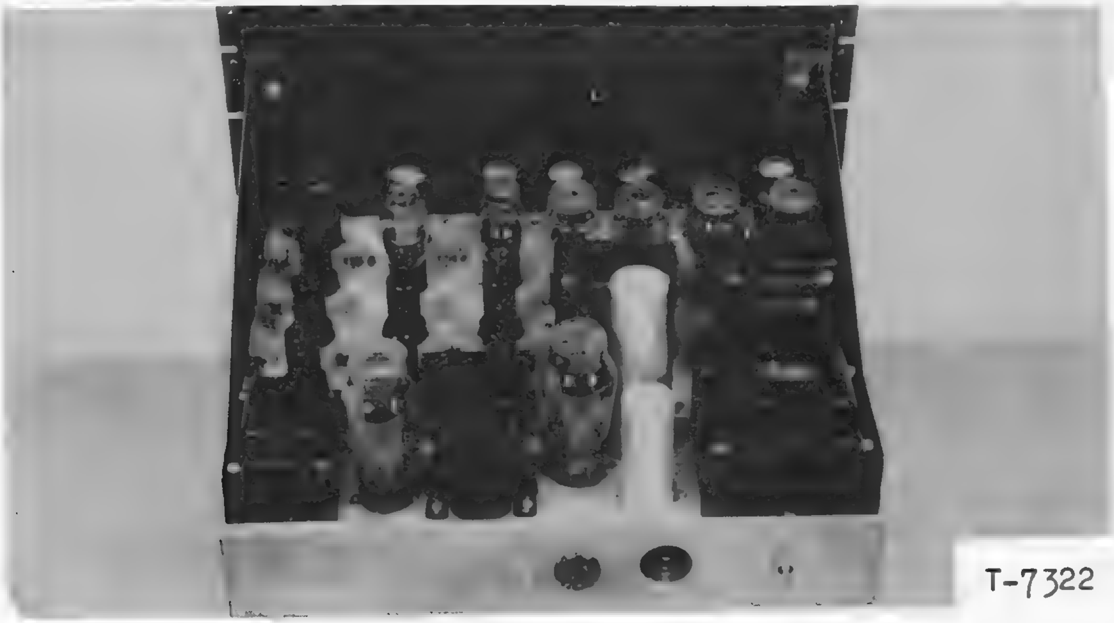
FIG 39 -- RANGE PHASE SHIFT AND DISCRIMINATOR UNIT



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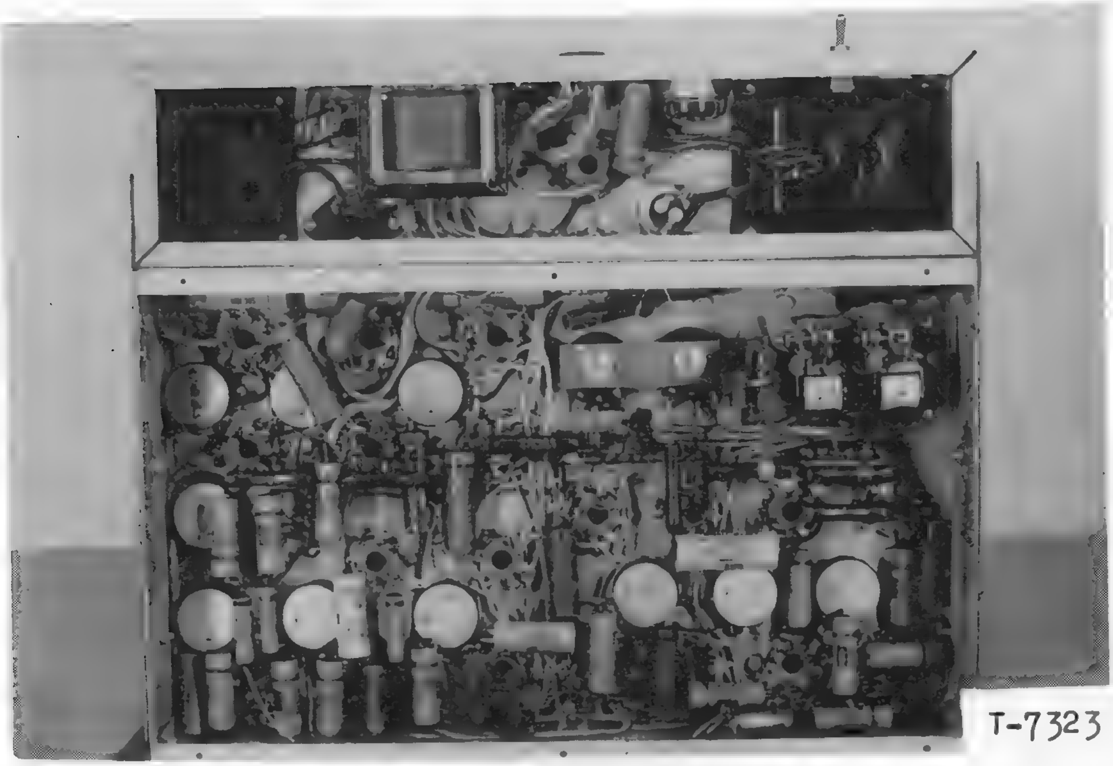

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FIG 40 -- MOTOR CONTROL UNIT OF RANGE SYSTEM

E. COMMAND CONTROL SYSTEM

THE COMMAND CONTROL DRIVE UNIT FOR USE WITH FLIGHT MISSILE NO. I HAS BEEN COMPLETED AND GIVEN OPERATIONAL TESTS IN THE LABORATORY. AMPLIFIERS WHICH MOUNT ON THE STABILIZATION PANEL ASSEMBLY FOR USE IN THE COMMAND CONTROL SYSTEM HAVE BEEN CONSTRUCTED AND ARE UNDERGOING OPERATIONAL TESTS.

AN ADDITIONAL COMMAND CONTROL DRIVE UNIT SIMILAR TO THE ONE CONSTRUCTED FOR THE NO. I FLIGHT MISSILE HAS BEEN COMPLETED AND INSTALLED IN A C-46 AIRPLANE FOR FLIGHT TESTING. IT IS INTENDED THAT A COMPLETE SYSTEM BE GIVEN OPERATIONAL TESTS IN FLIGHTS PRIOR TO SYSTEM INSTALLATION IN THE MISSILE.

TO PROVIDE FOR TESTING OF THE AIRBORNE FILTER AND SERVO CONTROL UNIT, A TEST MODEL SERVO RESPONSE UNIT HAS BEEN DESIGNED WHICH WILL FACILITATE FUTURE OPERATIONAL TESTS. THE ORIGINAL SERVO-RESPONSE UNIT HAD ONLY 30 TOTAL DEGREES MOVEMENT OF THE FOLLOW-UP PHASE-ROTATING-TRANSFORMER. SUBSEQUENT ALTERATIONS IN DESIGN OF THE TRANSFORMER ALLOW A GREATER PHASE ANGLE ROTATION, UP TO 180 DEGREES, FOR GENERATION OF COMMAND SIGNALS THEREBY REDUCING THE ERRORS DUE TO GRADUAL DRIFT RESULTING FROM THERMAL CHANGES, TUBE AGING AND OTHER CAUSES.

CONSTRUCTION OF THE AIRBORNE FILTER AND SERVO CONTROL UNIT HAS BEEN COMPLETED. THE UNIT HAS BEEN TESTED IN CONJUNCTION WITH THE SERVO RESPONSE SYSTEM AND PACKAGED FOR INSTALLATION IN THE MISSILE. FURTHER OPERATIONAL TESTS ARE PLANNED TO DETERMINE THE RATE OF RESPONSE AND THE RESPONSE AMPLITUDE FOR VARIOUS ERROR FREQUENCIES. DATA OBTAINED FROM THESE TESTS THEN WILL BE USED IN THE OVER-ALL STABILITY CONSIDERATIONS FOR CONTROLLED FLIGHT OF THE MISSILE.

THREE PRODUCTION TYPE SUPERHETERODYNE RECEIVERS TO BE USED IN THE COMMAND CONTROL SYSTEMS ARE PRESENTLY UNDER CONSTRUCTION. ONE HAS BEEN COMPLETED WITH THE REMAINING TWO IN THE EARLY STAGES OF ASSEMBLY.

A POWER SUPPLY FOR THE RECEIVER FOR THE SINGLE-CHANNEL AIRBORNE SYSTEM HAS BEEN COMPLETED WITH WORK PROCEEDING ON DESIGN AND CONSTRUCTION OF A BIAS POWER SUPPLY. THIS BIAS POWER SUPPLY IS BEING USED AS A MEANS OF REDUCING THE OVER-ALL POWER DRAIN BY THE RECEIVER. PREVIOUSLY, BIAS POWER HAD BEEN OBTAINED BY BLEEDER METHODS WHICH RESULTED IN HEAT LOSSES AND UNNECESSARY POWER CONSUMPTION.

SECTION VIII
TELEMETERING

A. TELEMETERING FOR THE TEST VEHICLE

THE TELEMETERING GROUND STATION INSTALLATION, WHICH HAS BEEN INSTALLED IN A TRUCK, CONSISTS OF THE FOLLOWING COMPONENTS: 2 RBF RECEIVERS; 2 DISCRIMINATOR UNITS, 3 RECORDING OSCILLOGRAPHS, A HIGH VOLTAGE POWER SUPPLY DESIGNED FOR THE OSCILLOGRAPHS, A 12-VOLT BATTERY FOR THE OSCILLOGRAPHS AND A 12-VOLT MG SET, 3 RAX RECEIVERS AND POWER SUPPLY, A DUMONT No. 241 OSCILLOSCOPE, A HEWLETT-PACKARD VTVM, AND OTHER ALLIED TEST EQUIPMENT.

A CATHODE-RAY TUBE COMPARATOR, FIGURE 41, WAS ADDED TO THE DISCRIMINATOR UNIT FOR CHECKING THE PRECISION OSCILLATOR AGAINST A TUNING FORK STANDARD AND FOR CHECKING FREQUENCY OF THE SUB-CARRIER OSCILLATORS.

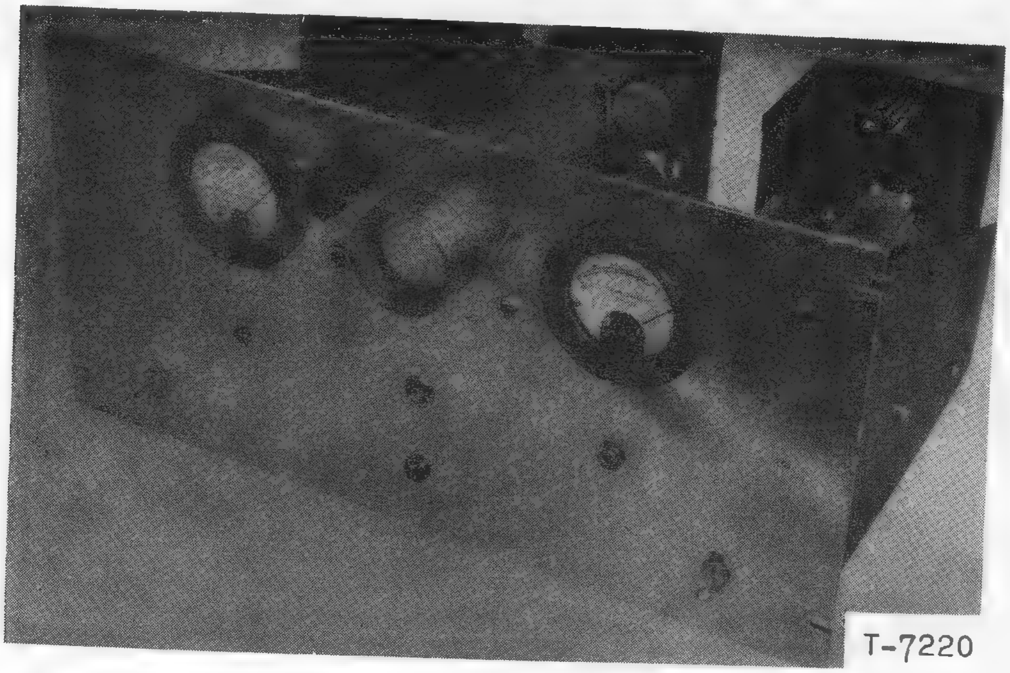

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FIG 41 -- CATHODE-RAY FREQUENCY COMPARATOR

The RBF receivers do not require the remote tuning equipment which normally is housed integrally with the receivers. When the receiver units are removed from the chassis provided by the manufacturer, new housing must be provided and controls relocated. Special cabinets have been constructed for use when these receivers, without remote tuning, are used in the telemetering system.

A telemetering transmitter-separation filter has been constructed as a spare to supplement the one currently in use with the telemetering system.

The dual commutator built for the Flight Test Missile will permit telemetering thirty-two channels of intelligence or identification, instead of twenty-two as will be done with the Static Test Missile. In conjunction with ten continuous channels, this dual unit will commutate two additional channels to provide eleven sub-channels each. Two of the twenty-two sub-channels are to be used for identification purposes only. The commutator is shown in Figure 42.

Production of sub-carrier oscillators is proceeding extremely well, 25 units have been completed in addition to the 18 previously completed for the static test missile. Principle advantage of the production type sub-carrier oscillators lies in the fact that preliminary checks and calibration can be made in a few hours, whereas the earlier experimental type oscillators frequently required from several days to a week to accomplish the same operations.

The telemetering installation at the static test site was made with the receiving antennas erected at a point approximately 400 feet south of the test tower. The operating station in the truck, linked to the antennas by coaxial cables, was placed at a protected location adjacent to the antennas. Views of the installation are shown in Figure 43. An intercommunication system afforded liaison with the firing personnel at the blockhouse. A photographic dark room was provided on the test site to permit rapid development of oscillographic records for on-the-spot analysis of missile functions.

Prior to firing of the missile for the "hot run", a telemetering check was made and the complete telemetering installation was functioning properly. However, because of damage to pickups and wiring during firing, the data obtained are of little value in analyzing the explosion.

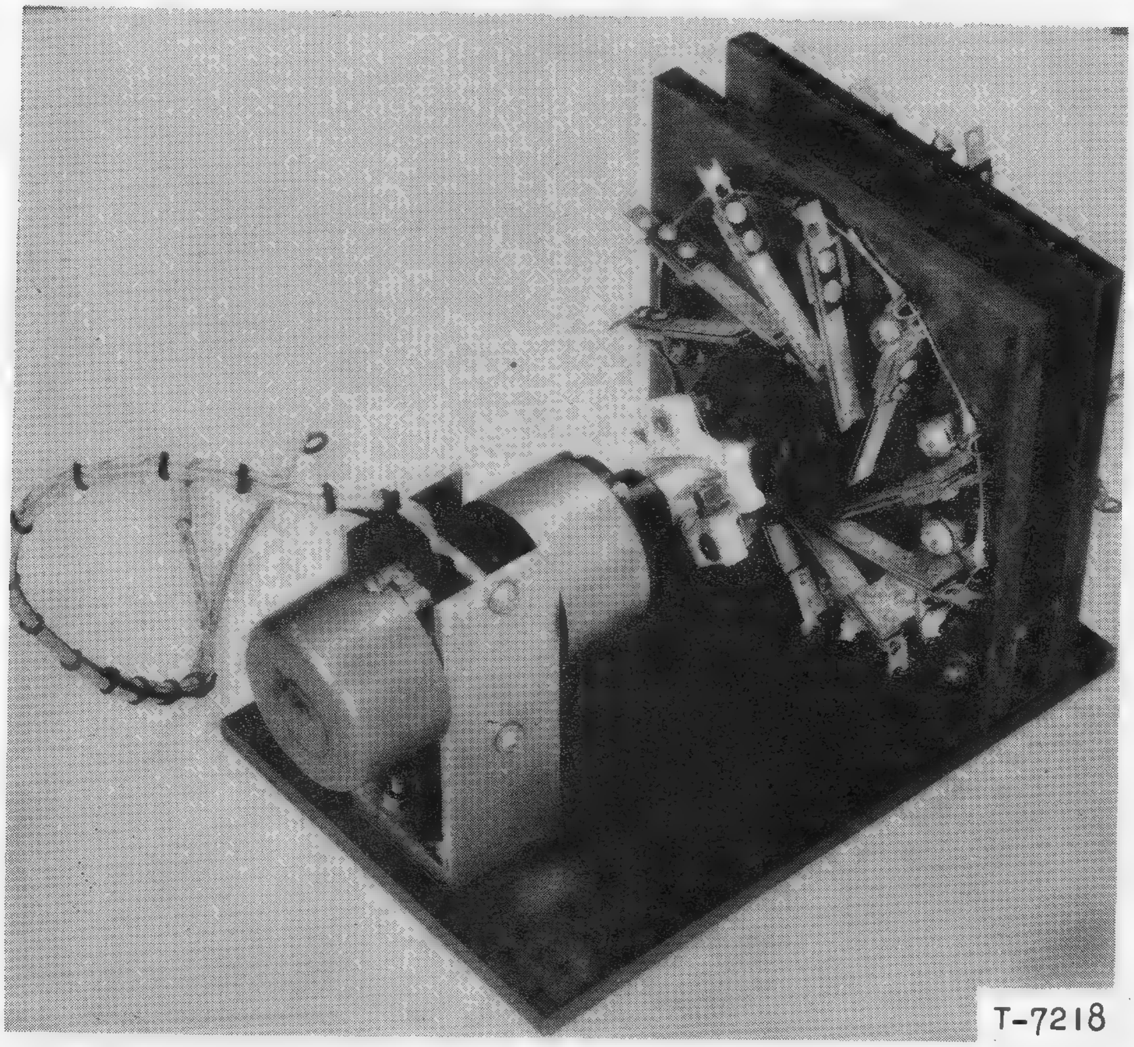

FIG 42 -- MOTOR DRIVEN DUAL COMMUTATOR

B. CAMERA LOCATOR BEACON

WORK ON THIS PROGRAM HAS BEEN SUSPENDED BECAUSE OF OTHER LABORATORY WORK OF A HIGHER PRIORITY. HOWEVER, DEVELOPMENT OF THIS UNIT HAS REACHED PRACTICAL COMPLETION WITH THE REMAINING WORK INVOLVING ONLY FINAL OPERATIONAL AND FIELD TESTS.

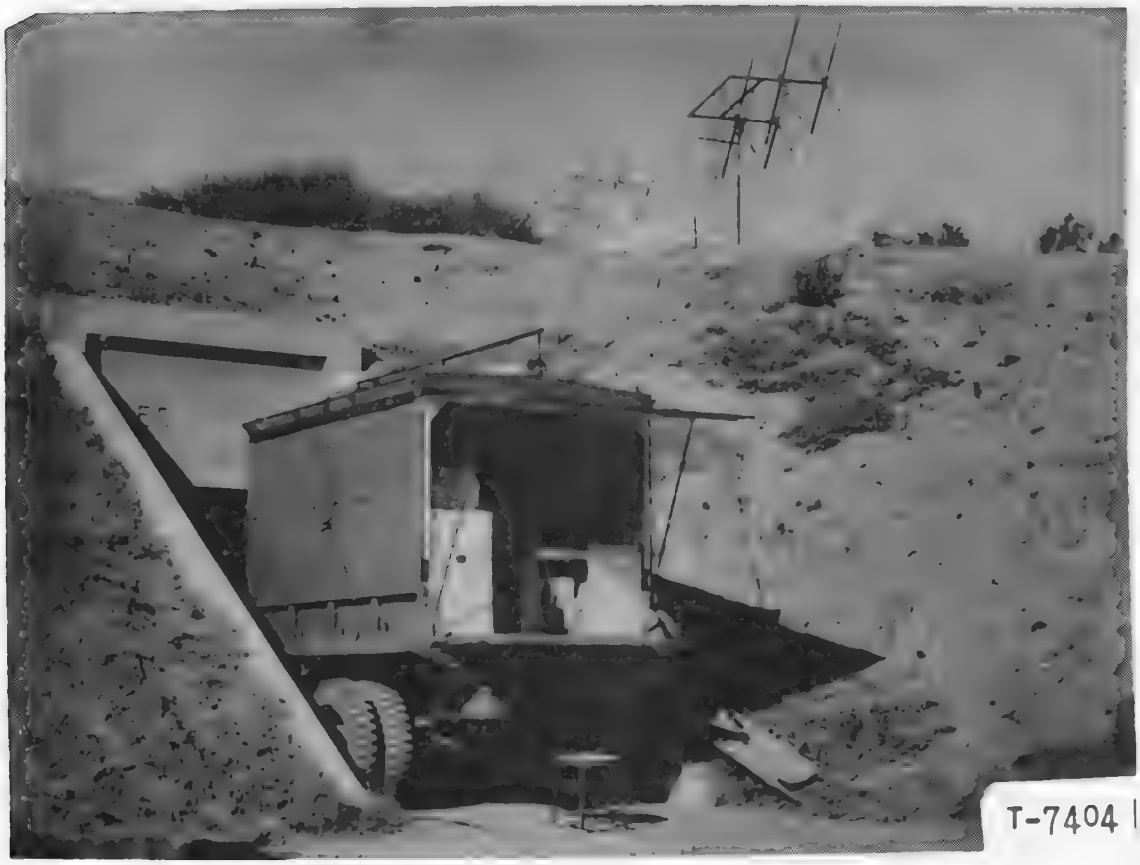

FIG 43 -- TELEMETERING TRUCK AND ANTENNA PLACEMENT AT POINT LOMA

C. TRACKING RADAR RELAY BEACON

DUE TO ASSIGNMENT OF ADDITIONAL FUNCTIONS TO THE RADAR BEACON, DESIGN OF THE SYSTEM CURRENTLY IS UNDERGOING REVISION TO ACCOMMODATE THE NEW REQUIREMENTS; VIZ: (1) AN INTERNAL POWER SUPPLY OPERATING FROM A 400-CYCLE, 115-VOLT A-C SOURCE, (2) A DETONATION TRIGGER FOR THE MISSILE DESTRUCTION CHARGE IN CASE THE DOPPLER TRIGGER FAILS TO FUNCTION PROPERLY, AND (3) INPUT FILTERS FOR THE ELIMINATION OF UNDESIRED S- AND X-BAND SIGNALS.

ADDITION OF THE INTERNAL POWER SUPPLY, DETONATION TRIGGER, AND ADDITIONAL FILTERS WILL NECESSITATE CONSIDERABLE REARRANGEMENT OF THE PRESENT COMPONENT POSITIONS IN ORDER TO ACCOMMODATE THE REVISED SYSTEM WITHIN THE MISSILE. A BREADBOARD LAYOUT OF THIS REVISION HAS BEEN COMPLETED. A BREADBOARD DESIGN OF THE INTERNAL POWER SUPPLY HAS BEEN COMPLETED WITH FURTHER DEVELOPMENT AWAITING CONSTRUCTION OF THE SPECIAL POWER TRANSFORMER AND CHOKE.

TO ELIMINATE THE UNWANTED S- AND X-BAND SIGNALS IT WAS FOUND NECESSARY TO USE TWO INPUT FILTERS. INITIALLY, A MODIFICATION OF THE CAVITY FILTER SUPPLIED WITH THE APN/19 EQUIPMENT WAS TESTED BUT IT PROVIDED INSUFFICIENT REJECTION OF X-BAND SIGNALS. A RESONANT CAVITY TUNED TO THE DESIRED FREQUENCY AND A LOW PASS "CONSTANT K" COAXIAL FILTER WERE THEN ADOPTED. SEVERAL OF THESE UNITS HAVE BEEN CONSTRUCTED AND THE FIRST OF THE UNITS INSTALLED IN THE STATIC TEST VEHICLE. TESTS OF THE UNITS SHOW THE RESONANT CAVITY PRESENTS 30 DB ATTENUATION TO S-BAND SIGNALS AT 15 MC OR GREATER OFF RESONANCE, AND THE LOW PASS FILTER ATTENUATES THE X-BAND SIGNALS GREATER THAN 75 DB. THE INSERTION LOSS OF EACH FILTER IS LESS THAN 0.1 DB.

THE ORIGINALLY DESIGNED RADAR BEACON, CONSISTING OF A RECEIVER AND A TRANSMITTER HOUSED IN A PRESSURIZED CONTAINER AND USING AN EXTERNAL POWER SUPPLY, HAS BEEN COMPLETED AND GIVEN EXTENSIVE ACCEPTANCE TESTS BEFORE INSTALLATION IN THE STATIC TEST MISSILE TOGETHER WITH THE INPUT FILTERS. THE UNIT IS CURRENTLY UNDERGOING REVISION TO INCORPORATE THE DETONATING TRIGGER CIRCUIT AND INTERNAL POWER SUPPLY.

COMPILATION AND EDITING OF AN INSTRUCTION BOOK TO BE USED IN CONJUNCTION WITH OPERATION AND MAINTENANCE OF THE RADAR BEACON HAS BEEN INAUGURATED.

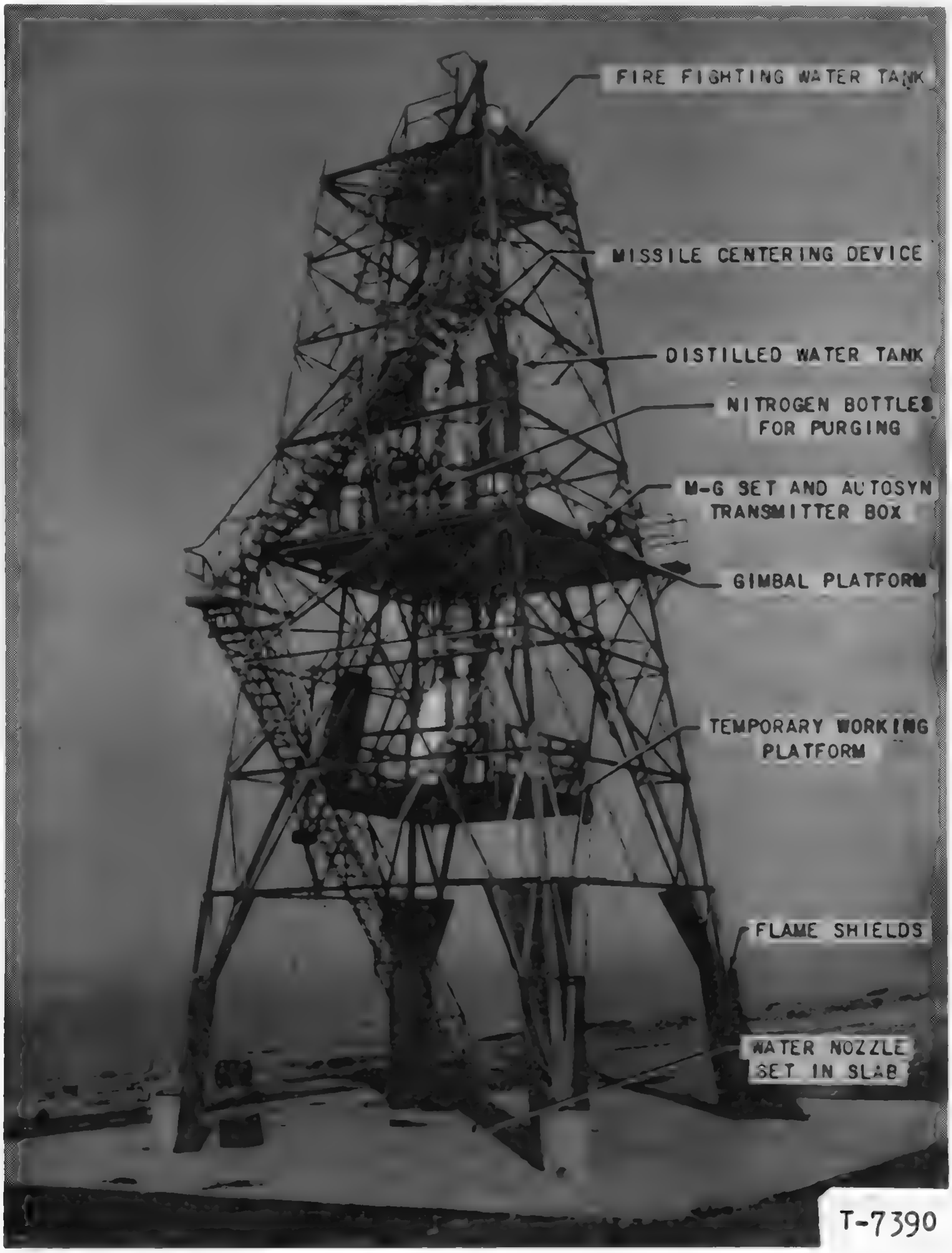

FIG 44 -- STATIC FIRING TEST STAND AT POINT LOMA, SAN DIEGO

SECTION IX
TEST OPERATIONS

A. STATIC FIRING REPORT

THIS REPORT PRESENTS A RECORD OF THE EVENTS THAT OCCURRED DURING THE FIRST FIRING OF THE MX-774 STATIC TEST VEHICLE AND THE RESULTS OBTAINED DURING THE FIRING OPERATIONS. THE STATIC TEST VEHICLE WAS INSTALLED IN THE CVAC STATIC TEST TOWER, FIGURE 44, LOCATED AT FORT ROSECRANS, POINT LOMA, CALIFORNIA. FIGURE 45 IS A PHOTOGRAPH OF THE AREA TAKEN EARLY IN THE TEST PREPARATIONS. THE FIRING TOOK PLACE ON 21 NOVEMBER 1947.

THE PURPOSE OF THE TEST WAS TO CHECK POWER PLANT, MISSILE STABILIZATION, AND TELEMETERING SYSTEMS FOR OPERATIONAL CHARACTERISTICS AND TO OBTAIN BASIC REFERENCE DATA, INCLUDING THRUST AND TEMPERATURE MEASUREMENTS, TO BE USED IN ANALYZING FLIGHT CHARACTERISTICS OF THE FREE-FLIGHT MISSILE.

FIG 45 -- STATIC FIRING SITE AT POINT LOMA IN EARLY PREPARATION

B. CONFIGURATION

The Static Test Vehicle was mounted in the gimbal rings of the static test tower. The gimbal rings were unlocked allowing movement of the missile in pitch, yaw, and roll directions. Movement in the pitch and yaw directions was restricted by thrust measuring device cables which were attached between the swivel hoist fitting on the nose of the missle and hydraulic cylinders located in the four corners of the tower. Snubbers were installed on the tower platform limiting the amount of roll to plus or minus ten degrees from a neutral position.

An emergency alcohol shut-off valve, remotely controlled from the blockhouse was installed at the alcohol tank outlet for use in case of fire. The external purge supply of nitrogen was piped from the nitrogen cascade system mounted on the tower platform. An electro-magnetic valve was installed in the line and controlled from the blockhouse. To manually regulate the thrust of the motors from the blockhouse, a Grove dome pressure regulator with pressure gauge was installed in the blockhouse and was plumbed to the second pressure regulator in the hydrogen-peroxide tank pressurizing line.

The large access door between the fins of the missile was left off to make possible visual observation and photographic recording of the power plant section of the missile during the run.

Blockhouse instrumentation with photo-recording, missile coverage by movie cameras located at advantageous locations, telemetering and stabilization equipment, and Brown recorder for temperature measurements were installed and thoroughly checked for operation prior to actual firing operations. Figure 46 shows the interior of the blockhouse.

The missile was fueled for full-time operation, however, the maximum limit of run was established at 35 seconds. Oxygen was pumped from a truck as shown in Figure 47.

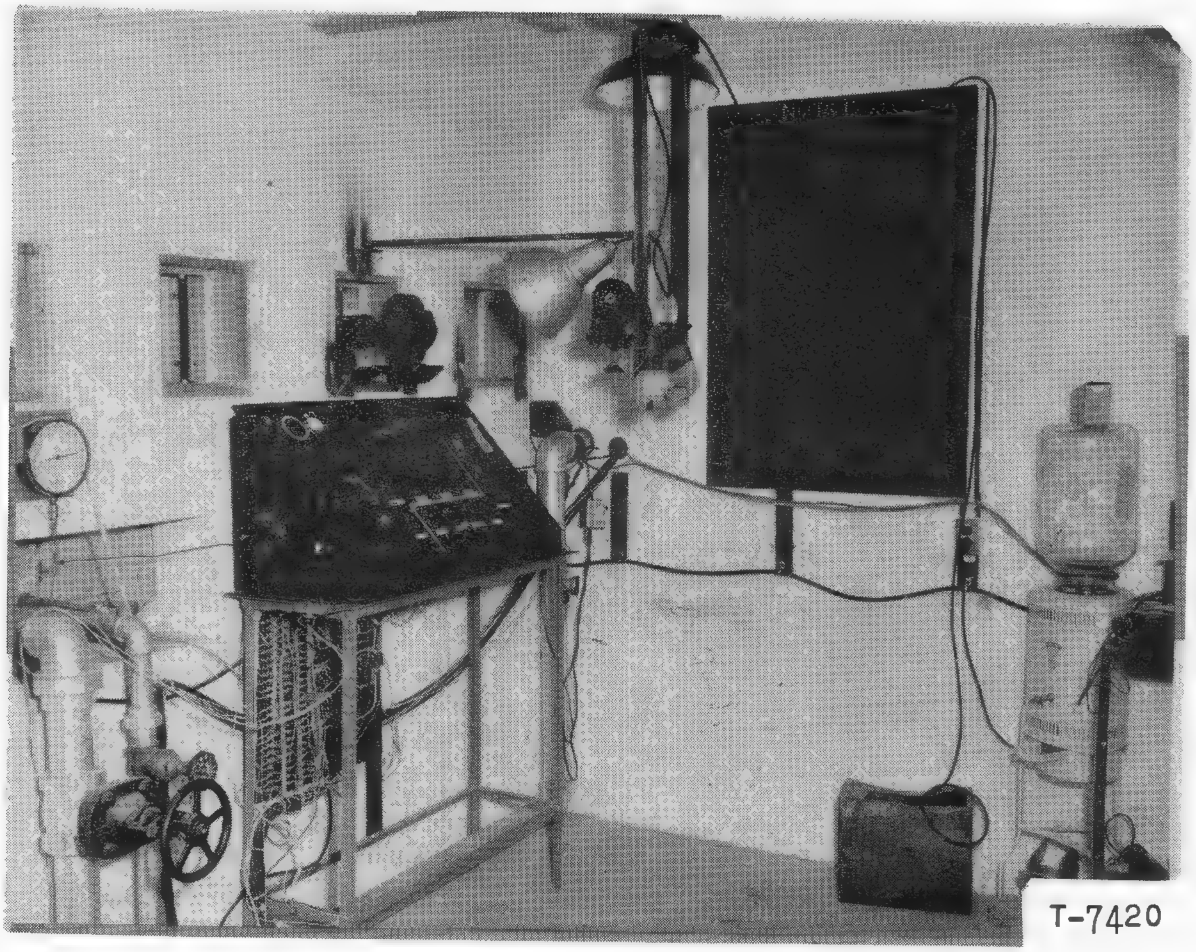

FIG 46 -- INTERIOR OF BLOCKHOUSE AT TEST SITE

C. PRELIMINARY TESTING

PRELIMINARY TO FIRING THE STATIC TEST VEHICLE MANY CHECKS WERE MADE OF THE VARIOUS SYSTEMS OF THE MISSILE. RESULTS OF THESE TESTS ARE AS FOLLOWS:

1. Power Plant Secondary Nitrogen System

IN ORDER TO PRESSURE CHECK THE SECONDARY NITROGEN SYSTEM FOR LEAKS THE FIRST STAGE PRESSURE REGULATOR WAS SET AT 400 PSI, AND THE HYDROGEN-PEROXIDE LINE AT THE STEAM GENERATORS WAS DISCONNECTED AND CAPPED. THE SECOND STAGE PRESSURE REGULATOR DID NOT HAVE A FIXED SETTING BUT COULD BE CONTROLLED FROM THE BLOCKHOUSE BY MEANS OF ITS DOME PRESSURE. THIS ARRANGEMENT WAS INTENDED TO PERMIT THE HYDROGEN PEROXIDE TANK PRESSURE AND THEREFORE THE TURBO-PUMP SPEED AND ROCKET THRUST

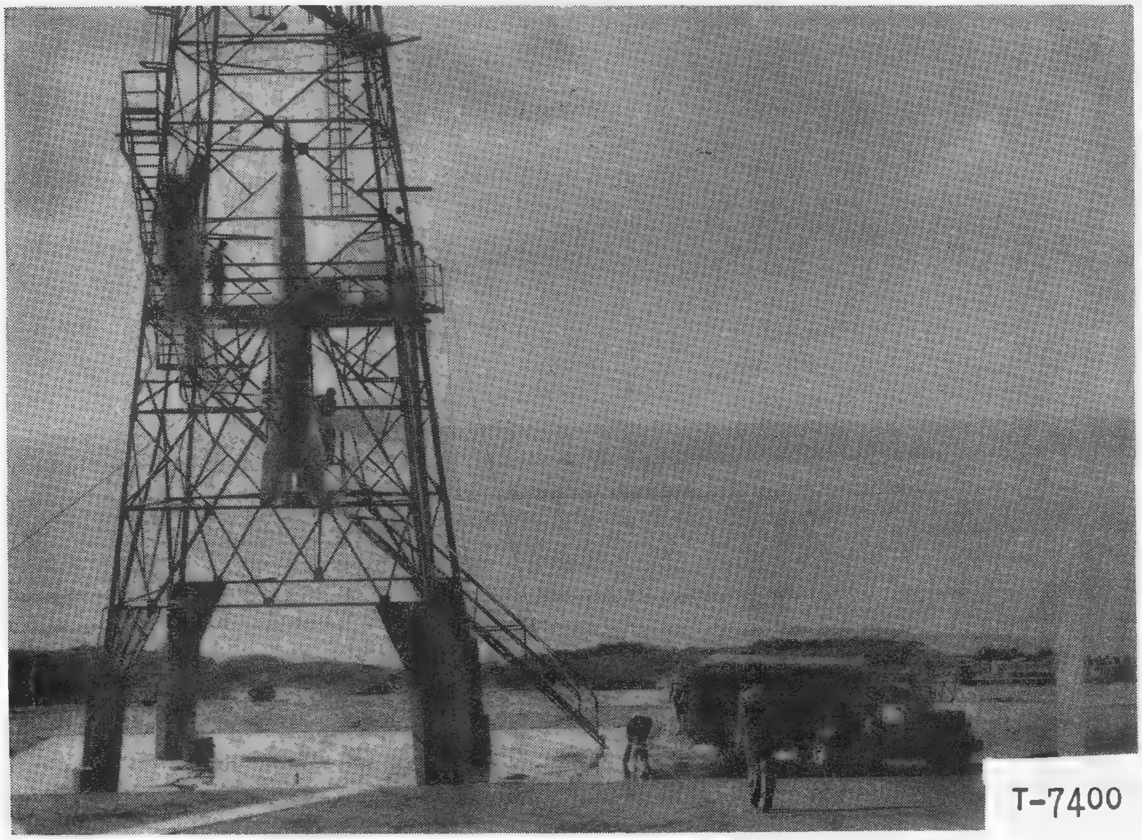

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FIG 47 -- OXYGEN FILLING OPERATION BEFORE FINAL CHECKOUT OF MISSILE

TO BE REGULATED BY THE OPERATOR IN THE BLOCKHOUSE. THE PEROXIDE TANK WAS FILLED WITH DISTILLED WATER TO MINIMIZE THE DANGER IN EVENT OF TANK RUPTURE. THE SECONDARY NITROGEN SYSTEM TANK WAS CHARGED TO 1000 PSI WITH NITROGEN GAS.

PROPER OPERATION OF THE H_2O_2 PRESSURIZING AND VENT VALVE WAS CHECKED BY ENERGIZING THE FIRING CIRCUIT. H_2O_2 TANK PRESSURE WAS CALIBRATED AGAINST THE DOME PRESSURE READ ON THE BLOCKHOUSE GAUGE.

DURING THIS TEST THE HYDROGEN PEROXIDE TANK FAILED UNDER PRESSURE. DUE TO LACK OF A PRODUCTION TANK CAPABLE OF WITHSTANDING THE REQUIRED PRESSURE, A SUBSTITUTE HYDROGEN-PEROXIDE TANK WAS MOUNTED ON THE OUTSIDE OF THE MISSILE. THIS CHANGE ALSO NECESSITATED INSTALLATION OF THE NITROGEN TANK ON THE OUTSIDE OF THE MISSILE.

2. POWER PLANT PROPELLANT SYSTEM

THE OXYGEN AND ALCOHOL SYSTEMS WERE PRESSURE CHECKED FOR LEAKS AND VENT VALVES CHECKED FOR OPENING PRESSURES. THE OXYGEN TANK VENT VALVE OPENED AT 47 PSI AND THE ALCOHOL TANK VENT VALVE OPENED AT 45.5 PSI. THE ALCOHOL PRESSURE SWITCH WAS SEPARATELY CHECKED OUT AND SET TO OPERATE AT 26.75 PSI.

AFTER ELIMINATING VARIOUS MINOR LEAKS THE PROPELLANT TANKS WERE CHECKED TO PRESSURIZE TO 32 PSI FROM THE MAIN NITROGEN TANK IN NORMAL OPERATIONAL SEQUENCE WHEN THE FIRING BUTTON WAS PRESSED.

3. POWER PLANT GENERAL

DURING THE PERIOD FOLLOWING RUPTURING OF THE HYDROGEN-PEROXIDE TANK, THE MISSILE FIN SECTION WAS REMOVED FROM THE TANK SECTION. BOTH TANKS WERE THOROUGHLY FLUSHED WITH CARBON TETRACHLORIDE TO REMOVE ANY ACCUMULATION OF FOREIGN MATTER, OILS, AND ALCOHOL.

THE HYDROGEN-PEROXIDE TANK AND LINES WERE PASSIVATED BY USING A CHROMIC ACID SOLUTION WHICH WAS POURED INTO THE TANK AND LEFT TO STAND OVERNIGHT.

THE DAY BEFORE ACTUAL FIRING, TROUBLE WAS ENCOUNTERED WITH THE SQUIB IGNITERS. PREPARATIONS FOR FIRING ON THIS DAY WERE COMPLETE; HOWEVER, CHECKS MADE ON THE SQUIB IGNITERS REVEALED THAT MANY SHOWED NO CONTINUITY ON AN OHMMETER. RESISTANCE ACROSS THE SQUIBS VARIED CONSIDERABLY FROM UNIT TO UNIT. A SELECTION OF SQUIBS INDICATING THE BEST CONTINUITY AND A MINIMUM OF RESISTANCE WERE CHECKED BY ATTEMPTING TO FIRE THEM. VERY FEW ACTUALLY FIRED. ANOTHER SELECTION WAS MADE TO PROVIDE IGNITORS FOR THE MISSILE. ALL SHOWED CONTINUITY BEFORE INSTALLATION, HOWEVER, A CONTINUITY CHECK MADE AFTER INSTALLATION REVEALED ONLY ONE OF THE FOUR INDICATED CONTINUITY. BASED ON THIS CONDITION THE HOT RUN SCHEDULED FOR 20 NOVEMBER WAS CALLED OFF.

SQUIB IGNITORS WERE TAKEN APART FOR INVESTIGATION. CAUSE OF THE TROUBLE WAS CORROSION AT ATTACHMENT OF THE CIRCUIT WIRE, WHICH WAS RUN THROUGH THE SQUIB, AND THE NICHROME FIRING WIRE. THIS CONNECTION WAS MADE UP BY WRAPPING THE NICHROME WIRE AROUND THE HEAVIER CIRCUIT WIRE. FOUR IGNITORS WERE REWORKED BY CLEANING AND MAKING MORE SECURE THE CONNECTION BETWEEN THESE TWO WIRES FOR THE FIRING WHICH TOOK PLACE ON THE FOLLOWING DAY.

4. STABILIZATION SYSTEM

PRELIMINARY TO THE HOT RUN A STABILIZATION SYSTEM COLD CHECK WAS MADE USING THE STABILIZATION SYSTEM TEST PANEL WITH FLIGHT PROGRAM TIMER INSTALLED. HYDRAULIC POWER TO MOVE THE ROCKET CYLINDERS WAS PROVIDED BY AN EXTERNAL ALCOHOL SUPPLY TANK WITH PUMP CONNECTED DIRECTLY TO THE MISSILE HYDRAULIC SYSTEM.

THE FLIGHT PROGRAM TIMER INDUCED SIGNALS TO:

(A) ZERO THE STABILIZATION SYSTEM

(B) GIVE STEP FUNCTION SIGNALS IN PITCH, YAW, AND ROLL

(C) RE-CHECK ZERO

(D) GIVE SINE WAVE SIGNALS IN ROLL, PITCH, AND YAW

(E) RE-CHECK ZERO

RESULTS OF THE COLD CHECK WERE VERY SATISFACTORY AS SHOWN BY SERVO RESPONSE CURVES TELEMETERED DURING COLD RUNS. THE ROCKETS MOVED THROUGH THE REQUIRED ANGULAR DISPLACEMENT AND RETURNED TO THEIR ZERO POSITIONS. IT WAS CONCLUDED THAT THE STABILIZATION SYSTEM EQUIPMENT WAS SATISFACTORY FOR THE HOT RUN.

5. TELEMETERING EQUIPMENT

DURING FIRST ATTEMPTS AT CHECKING OUT THE TELEMETERING EQUIPMENT CONSIDERABLE "HASH" OCCURRED IN THE MULTIVIBRATOR CIRCUITS. THIS REQUIRED REMOVAL OF THE TELEMETERING CAN FROM THE MISSILE FOR A DETAILED CHECK AND REPAIR. RING MODULATORS IN THE VALVE INPUT CHANNELS WERE REMOVED AS THEIR SENSITIVITY TO HEAT WAS THE IMMEDIATE CAUSE OF THE "HASH".

IT WAS LATER DISCOVERED THAT THE SOURCE OF TROUBLE WAS AN ERROR IN WIRING SUCH THAT THE POWER SUPPLY THAT WAS PULLING ALL ITS CURRENT FROM ONE PHASE OF A THREE PHASE CIRCUIT. WITH THIS ERROR CORRECTED, TELEMETERING CHECKS PRODUCED SATISFACTORY RESULTS. THE TELEMETERING EQUIPMENT WAS CONSIDERED READY FOR THE HOT RUN.

6. <u>INSTRUMENTATION</u>

CONSIDERABLE DIFFICULTY WAS EXPERIENCED IN GETTING THE AUTOSYN PRESSURE GAUGES TO OPERATE SATISFACTORILY. LOW VOLTAGE AND A MIX-UP BETWEEN MISSILE WIRING AND BLOCKHOUSE WIRING SEEMED TO BE THE MAJOR PROBLEM. THIS WAS CORRECTED PRIOR TO THE HOT RUN.

7. <u>DOPPLER</u>

THE SEQUENCE RELAY IN THE DOPPLER EQUIPMENT COULD NOT BE MADE TO OPERATE. CAUSE OF TROUBLE WAS NOT DETERMINED IN TIME FOR THE HOT RUN.

D. DESCRIPTION OF STATIC FIRING TEST

FIRING OF THE MISSILE TOOK PLACE ON 21 NOVEMBER 1947 AT APPROXIMATELY 11:30 A.M. THE MISSILE WAS FUELED FOR FULL-TIME OPERATION, HOWEVER, A MAXIMUM OPERATING TIME LIMIT WAS SET AT 35 SECONDS.

PERFORMANCE OF ALL OF THE SPECIALIZED DUTIES REQUIRED FOR FUELING, PREFIRING CHECKOFF, AND FIRING WERE ACCORDING TO SCHEDULE. THIS WAS MADE POSSIBLE BY PREVIOUS REHEARSALS OF ALL OPERATIONS AND PREPARATION FOR FIRING ON THE DAY PREVIOUS TO THE ACTUAL FIRING.

AT APPROXIMATELY TEN MINUTES BEFORE ACTUAL FIRING, THE EXTERNAL BATTERY SWITCH WAS THROWN ON, SETTING IN OPERATION ALL STABILIZATION AND TELEMETERING EQUIPMENT IN THE MISSILE FOR WARM UP AND CHECK OUT.

ALL CLEAR AND READY SIGNALS WERE RECEIVED IN THE BLOCKHOUSE FROM THE OBSERVATION, TELEMETERING, AND DOPPLER STATIONS. AT APPROXIMATELY TEN SECONDS BEFORE FIRING THE EXTERNAL PURGE (NITROGEN PURGE THROUGH THE OXYGEN PROPELLANT VALVE, LINES, AND ROCKETS) WAS SWITCHED ON AND HELD ON UP TO ONE-HALF SECOND BEFORE ACTUAL FIRING AT WHICH TIME THE PURGE WAS SWITCHED OFF. THE FIRING BUTTON WAS PUSHED AT ZERO TIME.

THE SEQUENCE OF EVENTS AS ANALYZED FROM THE BLOCKHOUSE INSTRUMENT PANEL RECORDING IS AS FOLLOWS:

TIME	OPERATION
ZERO	FIRING BUTTON PUSHED.
0 TO 0.9 SEC	PRESSURE IN H_2O_2 TANK INCREASED FROM 0 TO 210 PSI.
0.9 TO 1.4 SEC	PRESSURE IN BOTH PROPELLANT TANKS INCREASED TO OPERATING PRESSURE OF 20 PSI.
	THE H_2O_2 TANK PRESSURE CONTINUED TO INCREASE TO 340 PSI.
1.4 SEC	ALCOHOL PRESSURE SWITCH CLOSED CUTTING IN THE FIRING CIRCUIT AND OPENING ALL PROPELLANT VALVES.
1.4 TO 2.6 SEC	STEAM GENERATOR INLET AND OUTLET PRESSURES INCREASED AND STABILIZED AT APPROXIMATELY 250 AND 210 PSI, RESPECTIVELY.
	THE H_2O_2 TANK PRESSURE STABILIZED AT APPROXIMATELY 375 PSI.
2.6 SEC	CYLINDER NO. 1 CHAMBER PRESSURE INCREASED TO 50 PSI INDICATING IGNITION.
	CHAMBER PRESSURES IN CYLINDERS NO. 2, 3, AND 4 STILL AT ZERO.
	PROPELLANT PRESSURES AT INLET TO PROPELLANT VALVES HAD INCREASED TO 90 PSI FOR ALCOHOL AND 100 PSI FOR OXYGEN.
2.8 SEC	CHAMBER PRESSURE IN CYLINDER NO. 1, 65 PSI.
	CHAMBER PRESSURE IN CYLINDER NO. 2, 15 PSI.
	CHAMBER PRESSURE IN CYLINDERS NO. 2 AND 4, ZERO.
	PROPELLANT PRESSURES AT INLET TO PROPELLANT VALVES INCREASED TO 150 AND 130 PSI FOR ALCOHOL AND OXYGEN RESPECTIVELY.
	H_2O_2 TANK PRESSURE AND STEAM GENERATOR INLET AND OUTLET PRESSURES REMAINED AT 375, 250, AND 210 RESPECTIVELY.
2.9 SEC	OCCURRENCE OF EXPLOSION IS INDICATED BY THE FACT THAT ALL PRESSURE GAUGES GO ERRATIC WITH THE EXCEPTION OF THE STEAM GENERATOR INLET AND OUTLET PRESSURES. SOME GAUGES SHOWED VIOLENT OSCILLATIONS AND IT IS ASSUMED THAT ALL READINGS TAKEN FROM THIS POINT ON ARE ERRONEOUS.
6.3 SEC	CUT OFF SWITCHES WERE THROWN BY BOTH OPERATOR AND OBSERVER WHEN ABNORMAL FIRE WAS OBSERVED.

Following the explosion, Figure 48, and the start of fires, the alcohol shut-off valve at the alcohol tank outlet was closed, the tower water supply which was directed by nozzle into the power plant section through the opening between the fins was turned on, and the rocket water spray provided by the nozzle in the concrete apron was turned on, see Figure 48. These were all remotely controlled from the blockhouse. All electrical power to the tower was also cut off. The fires were immediately put out. However, dense clouds of vapor continued to issue from the missile power plant section. This was later discovered to have been caused by a break in the oxygen line at the propellant valve.

Observations made from the blockhouse during the run and results of investigations made following the run are summarized.

Apparently two cylinders fired, one of which seemed to fire normally and the other lagged. Explosions occurred in the barrels of two cylinders blowing the heads off. Fire immediately developed, the first flash of flame rising to the top of the tower. This was immediately followed by a fire within the power plant section of the missile. Fire also developed in the corner of the tower platform where the electrical wiring between the missile and conduit boxes accumulated.

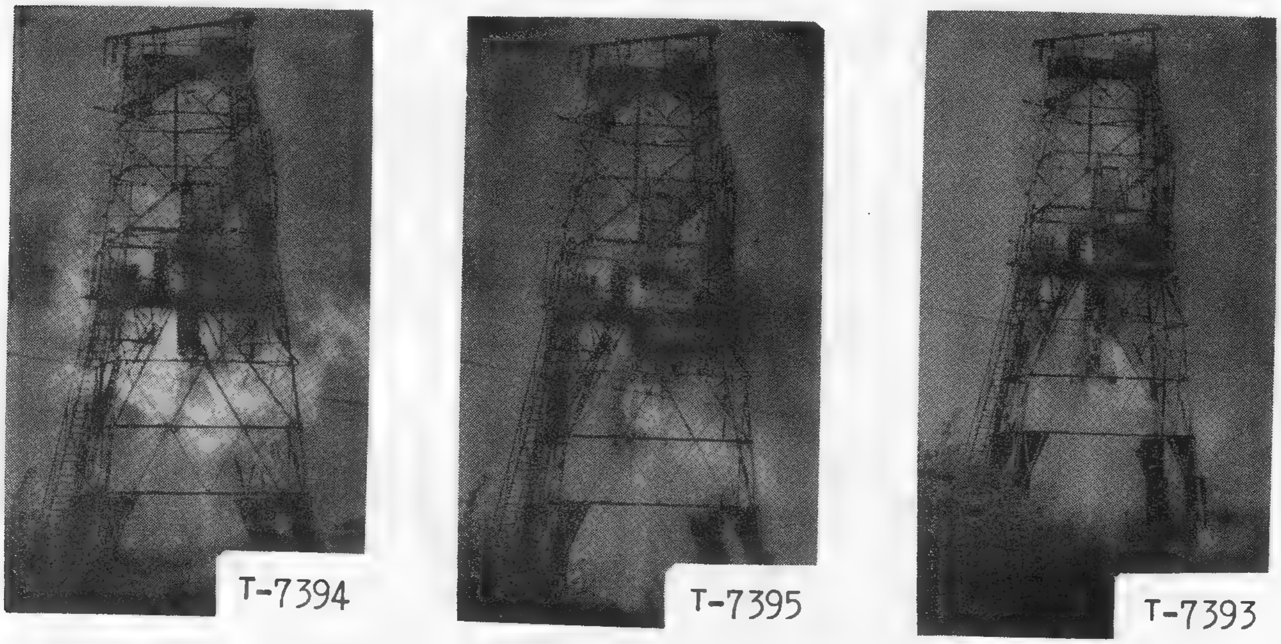

FIG 48 -- FLARE-UP FOLLOWED BY WATER-JET QUENCH OF FIRE IN THE ENGINE SECTION

E. DISCUSSION OF FIRING TEST RESULTS

1. POWER PLANT

ANALYSIS OF THE DATA SHEETS, FIGURE 49, TAKEN FROM INSTRUMENT PANEL RECORDINGS INDICATES THAT POWER PLANT OPERATION DURING THE PRESSURIZATION AND FIRING FUNCTIONS WAS SATISFACTORY. IMMEDIATELY FOLLOWING CLOSING OF THE FIRING CIRCUIT BY THE ALCOHOL PRESSURE SWITCH, TWO OF THE FOUR ROCKET CYLINDERS EXPLODED BLOWING THE CYLINDER HEADS CLEAR OF THE MISSILE.

CAUSE OF THE EXPLOSION CAN ONLY BE SURMIZED TO HAVE BEEN THE RESULT OF IGNITION OUTSIDE THE TWO CYLINDERS THAT EXPLODED. THE FOLLOWING CLUES ARE OFFERED TO SUBSTANTIATE THIS THEORY:

(A) THE WALLS OF THE TWO EXPLODED CYLINDERS ADJACENT TO THE HEADS WHICH BLEW OFF SHOW A VERY SEVERE BULGE INDICATING A LOCALIZED AREA OF VERY HIGH PRESSURE (FIGURE 19).

(B) ONE HEAD OF THE TWO EXPLODED CYLINDERS CONTAINED AN UNBURNED SQUIB IGNITOR. THIS WOULD INDICATE THAT THE FUEL IN THIS PARTICULAR CYLINDER MUST HAVE BEEN IGNITED FROM OUTSIDE THE CYLINDER, POSSIBLY FROM THE FLAME OF ANOTHER ROCKET.

(C) A FULL SECOND OF TIME ELAPSED BETWEEN OPENING OF THE PROPELLANT VALVES AND A RISE IN PRESSURE IN THE FIRST CYLINDER TO FIRE (INDICATING IGNITION). DURING THIS TIME A DENSE CLOUD OF UNBURNED PROPELLANTS WAS SEEN EMERGING FROM THE ROCKET NOZZLES.

IN THE LIGHT OF THESE CLUES A PLAUSIBLE EXPLANATION WAS CONSTRUCTED.

IT IS ASSUMED THAT THIS CLOUD WAS IGNITED THE INSTANT THE FIRST ROCKET FIRED.

INASMUCH AS THE OXYGEN PASSES DIRECTLY INTO THE CYLINDERS THROUGH THE INJECTOR NOZZLES THE INSTANT THE PROPELLANT VALVE IS OPENED BUT THE ALCOHOL MUST FILL AND THEN FLOW THROUGH THE ROCKET COOLING CHAMBER TO THE INJECTOR AND INTO THE CYLINDER, IT IS ASSUMED THAT THE DELAY CAUSED A LEAN MIXTURE IN THE PROPELLANT CLOUD INSIDE AND OUTSIDE OF THE ROCKETS CREATING FAVORABLE CONDITIONS FOR DETONATION.

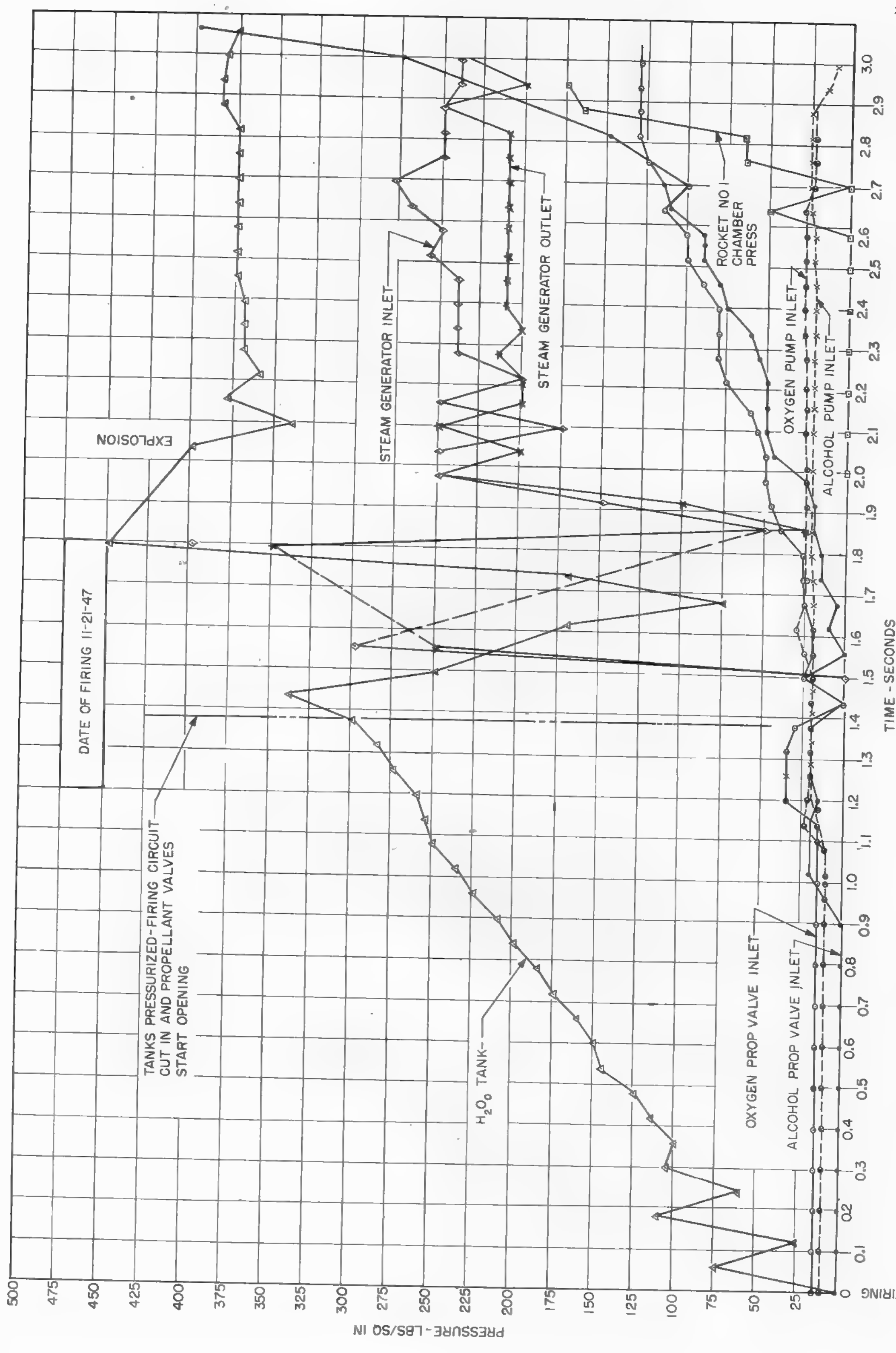

FIG 49 -- POWER PLANT PROPELLANT SYSTEM PRESSURES DURING FIRST THREE SECONDS OF STATIC FIRING TEST

Ignition of the rockets which failed occurred from the outside causing the flame front to advance into the nozzle of the rocket. The resulting increase in pressure in the cylinder caused the propellant mixture to rush out of the nozzle in a manner similar to normal ignition. However, in normal ignition, the unburned propellant would be pushed out of the nozzle ahead of the flame. In this case, all of the mixture was forced through the flame which caused its ignition.

This resulted in a greater quantity of mixture being burned in the cylinder than would normally occur causing excessive pressure and temperature. As the flame front progressed into the cylinder, pressure and temperature must have finally increased to a point where detonation occurred. This condition seemed to have occurred in both cylinders when the flame front had progressed to within approximately two inches of the injectors.

2. STABILIZATION

The stabilization system functioned normally from the time the firing switch was closed until the explosion occurred. Correct performance of the flight program switch and uncaging of the gyros were confirmed by visual inspection of the equipment after shut down. All four rockets apparently stabilized and remained in the zero position during the period between closing of the firing switch and the explosion. No records were obtained to corroborate this, however, inspection of the motion pictures showing the rockets indicate that the rockets remained in the zero position.

3. TELEMETERING, THRUST, AND TEMPERATURE

Telemetering equipment operated satisfactorily during the hot run and records were taken. The information obtained from the records was of little value in the analysis of the explosion, though the records did indicate that the equipment operated satisfactorily.

The thrust measuring equipment proved of no value in as much that a normal run did not occur. At the time of the explosion the missile lurched to the side and the instruments did not register. It is believed that the present method of measuring thrust may be satisfactory.

Temperature measurements were of no value due to the fact that up to the time of firing all temperatures were stabilized within a range of 0 to 15 degrees Centigrade and between fire and cutoff all temperatures were off the scale on the hot side.

SECTION X
PROJECTED WORK PROGRAM

WORK TO BE INITIATED, CONTINUED, OR COMPLETED DURING THE NEXT WORK PERIOD INCLUDES:

(1) COMPLETION OF THE PHASE CALIBRATOR, A SPARE TRANSMITTER, AND A SPARE RECEIVER FOR THE VHF POSITION TRACKER SYSTEM AND INAUGURATION OF FIELD AND FLIGHT TESTS OF TRACKER OPERATION.

(2) CONTINUATION OF DOPPLER RANGE AND SPEEDOMETER EXPERIMENTS.

(3) COMPLETION OF SUPERHETERODYNE RECEIVERS AND BIAS POWER SUPPLY FOR THE COMMAND CONTROL SYSTEM AND CONTINUATION OF OPERATION CHECKS.

(4) CONTINUATION OF STABILIZATION PANEL ASSEMBLY FOR THE FLIGHT TEST VEHICLE.

(5) CONTINUATION OF TESTS ON INTEGRAL CORRECTION AND RESPONSE UNIT AND ON THE SIMULATOR.

(6) COMPLETION AND RELEASE OF REPORTS ON HYPERGRID NAVIGATION, HYPERGRID TEST SYSTEM, AND AUTOMATIC GUIDANCE - PEAK COMPARATOR AMPLIFIER.

(7) COMPLETION OF ANALYSIS ON WARHEAD SURFACE HEATING.

(8) REPAIR OF DAMAGE TO STATIC TEST MISSILE AND TO POINT LOMA TOWER AND REMOUNT OF MISSILE IN THE TOWER.

(9) COMMENCEMENT OF C-46 FLIGHT TESTS ON GUIDANCE EQUIPMENT.

TECHNICAL DATA

REPORT NUMBER	TITLE
A. ACTIVITY REPORTS	
1496-1	Ground-to-Ground Missile Progress, May 1946
1496-2	Ground-to-Ground Missile Progress, Jun 1946
1496-3	Ground-to-Ground Missile Progress, Jul 1946
1496-4	Ground-to-Ground Missile Progress, 10 Jul - 10 Aug 1946
1496-5	Ground-to-Ground Missile Progress, 10 Aug - 10 Sep 1946
1496-6	Ground-to-Ground Missile Progress, 10 Sep - 10 Oct 1946
1496-7	Ground-to-Ground Missile Progress, 10 Oct - 10 Nov 1946
1496-8	Ground-to-Ground Missile Progress, 10 Nov - 10 Dec 1946
1496-9	Ground-to-Ground Missile Progress, 10 Dec 1946 - 10 Jan 1947
1496-10	Ground-to-Ground Missile Progress, 10 Jan - 1 Mar 1947
1496-11	Ground-to-Ground Missile Progress for Mar - Apr 1947
1496-12	Ground-to-Ground Missile Progress for May - Jun 1947
1496-13	Ground-to-Ground Missile Activity for Jul - Aug 1947
B. INTERNAL REPORTS	
DEVF-275-805	Power Plant - Oxygen Tank - Static and Pressure Test of
DEVF-275-824	Fin - Army Test Vehicle - Static Test of
DEVF-301-126	Fuel System MX-774 Supersonic Test Vehicle Tests of Pressurization by Vaporization of Liquid Oxygen
DEVF-346-16	Study of Propellants of High Specific Impulse; 25 Sep 1946

REPORT NUMBER	TITLE
DEVF-346-17	CALCULATION OF SKIN TEMPERATURES OF THE SUPERSONIC TEST VEHICLE AND THE SUPERSONIC GROUND-TO-GROUND MISSILE; 25 Nov 1946
DEVF-4002	PRELIMINARY ANALYSIS OF VIBRATING REED GYROSCOPE; 21 JUN 1946
DEVF-4008	PROGRESS REPORT OF PRECISION GYROSCOPE; 29 JUL 1946
DEVF-4009	POSITIVE ION ACCELEROMETER - ESTIMATES OF OUTPUT VOLTAGE AND REQUIRED XENON PRESSURE; 13 AUG 1946
DEVF-4012	THE USE OF THE CYCLOTRON PRINCIPLE FOR INCREASING ACCELEROMETER TRANSIT; 22 AUG 1946
DEVF-4013	MAGNETIC NAVIGATION INVESTIGATION; 28 AUG 1946
DEVF-4014	MISSILE VELOCITY DETERMINATIONS FROM DOPPLER EFFECT; 23 AUG 1946
DEVF-4016	THREE-GROUND-STATION RADAR NAVIGATIONAL SYSTEMS FOR A PROJECTILE-TYPE MISSILE; 6 SEP 1946
DEVF-4018	IONOSPHERE REFRACTION ERROR ESTIMATE OF SIGHTING ERROR AND INCREASE IN PHASE VELOCITY; 5 NOV 1946
DEVF-4019	INVESTIGATION OF MAXIMUM ERROR OF THREE-GROUND STATION NAVIGATIONAL SYSTEM; 8 NOV 1946
DEVF-4024	PROPORTIONAL RADIO CONTROL; 10 SEP 1946
DEVF-4025	TRANSMITTING ANTENNAS FOR VHF HYPERBOLIC GRID GUIDANCE CONTROL SYSTEM; 15 OCT 1946
DEVF-4032	SLAVE STATION RECEIVING ANTENNA FOR HYPERBOLIC GRID GUIDANCE SYSTEM; 14 NOV 1946
DEVF-4036	ATMOSPHERIC REFRACTION ERROR ESTIMATE OF SIGHTING ERROR AND CHANGES IN PHASE VELOCITY; 3 FEB 1947
DEVF-4038	A PRECISION MISSILE TRACKING SYSTEM; 30 DEC 1946
DEVF-4039	STABILIZATION AND CONTROL SYSTEMS FOR MISSILE MX-774; 31 DEC 1946
DEVF-4040	ANTENNA SYSTEM FOR VHF DOPPLER SPEED INDICATOR; 3 JAN 1947
DEVF-4042	A STUDY OF WAVE PROPAGATION CHARACTERISTICS AS AFFECTING THE HYPERGRID NAVIGATION SYSTEM; 2 JUL 1947

REPORT NUMBER	TITLE
DEVF-4043	Airborne Receiving Antennas for VHF Hyperbolic Grid Guidance System; 5 Mar 1947
DEVF-4045	Airborne Doppler Speed Indicator Ground Station Antennas; 17 Mar 1947
DEVF-4052	Selection of Guidance System for MX-774 Missile; 11 Jun 1947
DEVF-4054	Low Frequency, Long Range Navigation System Proposal; 14 Aug 1947
DEVF-4055	VHF Airborne Antennas for Army Test Vehicle; 3 Jul 1947
DEVF-5008	Roll Damping Test of 1/8-Scale Model MX-774 Single Stage Supersonic Test Vehicle in CVAC 4-Foot Wind Tunnel
DEVF-5011-1	Structural Design Criteria Single Stage Test Vehicle; 15 Nov 1947
DEVF-5013	Three Component Subsonic Wind Tunnel Test of 1/8-Scale Model of MX-774 in CVAC 4-Foot Wing Tunnel; 14 Feb 1947
DEVF-5015	Tab Hinge Moments of 1/2-Scale Model of V-2 Type Fin for MX-774 in CVAC 4-Foot Wind Tunnel; 28 Feb 1947
DEVF-5019	Three Component Subsonic Wind Tunnel Tests of 1/8-Scale Model MX-774; 7 Mar 1947
DEVF-5029	Project MX-774 Status to 1 May 1947; 2 May 1947
DEVF-5033	Power Plant Model Tests, MX-774 Test Stand; 6 Jun 1947
DEVF-5034	Tactical Missile Fuel Oxidizer Location Study
ZA-6002-001	Proposal for Air-Launched Sounding Rocket
ZA-6002-002	Final Aerodynamic Report on the Single Stage Test Vehicle (MX-774)
ZK-6002-001	Review of Pressurization System Single Stage Vehicle
ZM-360	Magnetic Guidance for Long Range Missiles; 8 Aug 1947
ZN-001	A Study of Range Measurement Systems; 25 Aug 1947
ZN-6002-003	Telemetering Receiving Antenna Array for Army Test Vehicle; 7 Aug 1947

REF.	REPORT NUMBER	TITLE
	ZN-6002-005	IONIC ACCELEROMETER EXPERIMENTS; 15 AUG 1947
	ZN-6002-006	AUTOMATIC MAGNETIC GUIDANCE
	ZN-6002-013	A PERIODIC COMPUTER FOR THE AUTOMATIC CONTROL OF FLIGHT PATH; 20 OCT 1947
	ZN-6002-014	MISSILE ANTENNA DEVELOPMENT FOR PHASE COMPARISON POSITION TRACKING SYSTEM; 22 OCT 1947
	56ZR-6002-001	MX-774 TRAJECTORY STUDIES SUMMARY REPORT

C. REFERENCES OF THIS ACTIVITY REPORT

1	DEVF-5011-1	STRUCTURAL DESIGN CRITERIA SINGLE STAGE TEST VEHICLE; 15 NOV 1947
2	DEVF-1496-13	GROUND-TO-GROUND MISSILE ACTIVITY FOR JUL - AUG 1947
3	DEVF-346-17	CALCULATION OF SKIN TEMPERATURES OF THE SUPERSONIC TEST VEHICLE AND THE SUPERSONIC GROUND-TO-GROUND MISSILE; 25 NOV 1946
4		LEWIS AND VON ELBE; JOURNAL OF AMERICAN CHEMICAL SOCIETY; V57--P612 (1935)
5		B. F. DODGE; CHEMICAL ENGINEERING THERMODYNAMICS, P372; NEW YORK 1944.
6	ZN-6002-014	MISSILE ANTENNA DEVELOPMENT FOR PHASE COMPARISON POSITION TRACKING SYSTEM; 22 OCT 1947

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